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SUPPLEMENT 3

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

APOLLO 14 MISSION REPORT

SUPPLEMENT 3

SERVICE PROPULSION SYSTEM FINAL FLIGHT EVALUATION

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MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
MAY 1972

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PROJECT TECHNICAL REPORT

APOLLO 14
CSM 110

SERVICE PROPULSION SYSTEM
FINAL FLIGHT EVALUATION

Follow-on Contract to NAS 9-8166 September 1971

Prepared for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

Prepared by
Propulsion Systems Section
Chemical and Mechanical Systems Department

TRW
SYSTEMS GROUP

MSC-04112
Supplement 3

APOLLO 14 MISSION REPORT

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SERVICE PROPULSION SYSTEM
FINAL FLIGHT EVALUATION

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1. PURPOSE AND SCOPE

The purpose of this report is to present the results of the postflight analysis of the Service Propulsion System (SPS) performance during the Apollo 14 Mission. This report is a supplement to the Apollo 14 Mission Report. The primary objective of the analysis was to determine the steady-state performance of the SPS under the environmental conditions of actual space flight.

This report covers the additional analyses performed following the compilation of Reference 1. The following items are the major additions and changes to the results reported in Reference 1:

1) The steady-state performance as determined from analysis of the third and seventh burns is presented.
2) The analysis techniques, problems and assumptions are discussed.
3) The flight analysis results are compared to the preflight predicted performance.
4) The propellant utilization and gaging system (PUGS) operation is evaluated in greater detail.
5) The pressurization system performance is discussed.
6) The transient data and performance are included.
7) The propellant consumption estimates are revised.

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2.0 SUMMARY

CSM 110 SPS performance for the Apollo 14 Mission was evaluated and found to be satisfactory. The SPS mission duty cycle consisted of seven firings for a total duration of 574.29 seconds.

SPS steady-state performance was determined primarily from the analyses of the third (LOI-1) and seventh (TEI) burns. It was determined from these analyses that the engine fuel resistance was approximately 6.7% less than its acceptance test value. This compares well with the mean fuel resistance bias of -5.5% determined from postflight analysis of Apollo 9, 10, 11, and 12 (Reference 9) and used in the Apollo 14 preflight analysis.

Average standard inlet condition engine performance values for the two burns analyzed are as follows: thrust - 20736 pounds; specific impulse - 314.0 seconds; and propellant mixture ratio - 1.557 units. These values are 0.4% greater, 0.1% greater, and 0% different, respectively, than corresponding values computed from the preflight engine model. Individual standard inlet condition performance for the two burns showed good agreement with differences of only 10 pounds for thrust, 0.2 seconds for specific impulse and no difference for mixture ratio.

Operation of the Propellant Utilization and Gaging System (PUGS) was satisfactory throughout the mission. The PUGS mode selection switch was set in the normal position for all SPS burns; therefore, only the primary system data were available. The propellant utilization (PU) valve was in the increase position at launch to compensate for the negative engine mixture ratio bias (fuel resistance bias) expected from past flights. By utilizing the PU valve the crew achieved excellent propellant management with the unbalance at the end of the seventh burn being only 40 lbm.

3. INTRODUCTION

The Apollo 14 Mission was the fourteenth in a series of flights using Apollo flight hardware and was the third lunar landing mission of the Apollo Program. It included the seventh flight test and the sixth manned flight of the Lunar Module (LM), the eighth manned flight of the Block II Command and Service Module (CSM), and was the seventh manned flight using a Saturn V launch vehicle. The objectives of the mission were to investigate the lunar surface near a preselected point in the Fra Mauro formation, deploy and activate an Apollo lunar surface experiments package, further develop man's capability to work in the lunar environment, and obtain photographs of candidate exploration sites. Combined spacecraft functions included Command Module docking with the LM, spacecraft separation from the launch vehicle, seven SPS firings, one Descent Propulsion System (DPS) firing, two Ascent Propulsion System (APS) firings, a rendezvous and docking, and a SPS Transearth Injection (TEI) firing.

The space vehicle was launched from the Kennedy Space Center (KSC) at 4:03:02 PM E.S.T. on 31 January 1971. Because of unsatisfactory weather conditions at the planned time of launch, a launch delay (about 40 minutes) was experienced for the first time in the Apollo program. The clock on-board the spacecraft was changed at about 54 hours after launch by adding 40 minutes and 2.90 seconds. Had the clock update not been performed, indications of elapsed time in the crew's data file would have been in error by the amount of the delay in lift-off, since the midcourse corrections were targeted to achieve the prelaunch - desired lunar orbit insertion time. All times herein are in elapsed time from lift-off (AET). The S-IVB stage was restarted approximately 2-1/2 hours after launch for the Trans-Lunar

Injection (TLI) Maneuver. During transposition and docking, about 5 hours after the TLI Maneuver, six attempts were required to achieve docking because of mechanical difficulties.

There were seven SPS burns during the mission, with a total duration of 574.29 seconds. The first SPS burn was the Hybrid Maneuver performed approximately 30-1/2 hours after liftoff. At approximately 77 hours the second docked SPS firing, a mid-course correction, was performed. The third, and longest, SPS burn was the Lunar Orbit Insertion (LOI) Maneuver conducted at about 82 hours. Approximately 4 hours later, the fourth SPS burn, the Descent Orbit Insertion (DOI) Maneuver was performed. The lunar module was undocked from the command module at about 103-3/4 hours. At about 105-1/6 hours the fifth SPS burn, the circularization (CIRC) Maneuver was performed. Approximately 3 hours later the Powered Descent Initiation (PDI) was performed by the DPS, terminating in the lunar landing at 108:15:09.3 hours. At about 117-1/2 hours the sixth SPS burn, the Lunar Orbit Plane Change (LOPC) was performed. The Lunar Liftoff burn was performed by the APS at approximately 142 hours, and a direct rendezvous was performed and the command-module - active docking operations were normal. Approximately 7 hours later the seventh SPS burn, the Trans-Earth Injection (TEI) Maneuver, was performed.

The actual ignition time, burn duration and velocity gain for each of the seven firings are contained in Table 1.

The Apollo 14 Mission utilized CSM 110 which was equipped with SPS Engine S/N 63 (Injector S/N 121). The engine configuration and expected performance characteristics (Reference 2) are contained in Table 2.

The SPS engine was started in the single bore engine valve mode on all seven burns. The second and fifth burns were conducted completely in the single bore mode. For the remaining burns the other bore was opened 1 to 4 seconds after ignition.

The first three SPS firings were no-ullage starts, while the remaining burns were preceded by +X Service Module (SM) reaction control system translation maneuvers to ensure SPS propellant settling. All SPS firings were conducted under automatic control.

4. STEADY-STATE PERFORMANCE ANALYSIS

Analysis Technique

The major analysis effort for this report was concentrated on determining the steady-state performance of the SPS during the third and seventh burns. The remaining five burns were of insufficient duration to warrant detailed performance analysis. The performance analysis was accomplished with the aid of the Apollo Propulsion Analysis Program (PAP) which utilizes a minimum variance technique to "best" correlate the available flight and ground test data. The program embodies error models for the various flight and ground test data that are used as inputs, and by statistical and iterative methods arrives at estimations of the system performance history, propellant weights and spacecraft weight which "best" (minimum-variance sense) reconcile the available data.

Analysis Description

The steady-state performance during the third burn was derived from the PAP analysis of a 324-second segment of the burn. The segment analyzed began approximately 24 seconds following ignition (FS-1). The first 24 seconds of the burn were not included, in order to minimize any errors resulting from data filtering spans which include transient data, and because PUGS data near the start of the burn are erroneous. The time segment analyzed was terminated approximately 23 seconds prior to SPS shutdown (FS-2) to avoid shutdown transients. The burn segment included four PU valve movements, and propellant crossover (storage tank depletion) which occurred about 243 seconds after ignition. The seventh burn steady-state performance was derived from the PAP analysis of a 102 second segment of the burn. The initial 33 seconds of the burn were excluded from the segment to avoid inclusion of data from the start transient, and the PU valve movement which occurred early in the burn. The segment was terminated approximately 15 seconds prior to engine cutoff in order to exclude shutdown transient data. The steady-state performance analyses of both burns utilized data from the flight measurements listed in Table 3.

The initial estimated spacecraft damp weight (total spacecraft minus SPS propellant) at ignition of the third burn was 57016 lbm. The initial estimated damp weight at ignition of the seventh burn was 22765 lbm. Both values were based on the postflight weight analysis given in Reference 3.

The initial estimates of the SPS propellants onboard at the beginning of the time segment analyzed for the third burn were extrapolated from the loaded propellant weights presented in Section 5. The initial propellant estimates for the time segment analyzed for the seventh burn were extrapolated from the computed propellants remaining at the end of the time segment analyzed for the third burn. All extrapolations of propellant masses used to establish the initial estimates for a given simulation were performed in an iterative manner using derived flowrates and propellant masses from preceding simulations to ensure that the derived propellant mass history was consistent between the two burns analyzed.

The SPS engine thrust chamber throat area was input to the program as a function of time from ignition for each burn. The assumed throat area time history used in the analysis is shown in Figure 1 and was based on the characterization presented in Reference 2.

The SPS propellant densities used in the analysis were calculated from propellant sample specific gravity data obtained from KSC, flight propellant temperature data, and flight interface pressures. The temperatures used were based on data from feed-system and engine feedline temperature measurements and were input to the program as functions of time. During steady-state operation, it was assumed that respective tank bulk temperatures and engine interface temperatures were equal for both oxidizer and fuel.

The PAP simulations were performed using an "interface pressure driven" SPS model. Simply stated, this model utilizes input oxidizer and fuel engine interface pressure values, as functions of time, for the starting points in computing the pressures and flowrates throughout the system. The input interface pressures used are generally the filtered data from the flight interface pressure measurements. The program is free to bias the input pressures, if so required, to achieve a minimum variance solution, but the version used (Linear Model 0) is essentially constrained to follow the shape of the input interface pressure profiles. The shapes of the interface pressure profiles, in turn, strongly influence the computed thrust shape, and, therefore, the calculated acceleration shape. The initial simulations of both burns, using the filtered interface pressure data, yielded minor computed acceleration shape errors. Analysis of the acceleration shape errors indicated that the filtered oxidizer and fuel interface pressure data were slightly in error. Shape errors in the filtered data are not unusual and are primarily the result of the PCM quantization of the raw data, which for the interface pressures is approximately 1.2 psi/PCM count. By utilizing the noise-in-the-state version (Linear Model 2) of the program, it was possible to derive corrections to the filtered interface pressure data which significantly improved the overall data match. The corrections, which were all less than 1.0 psi, were then input to the Linear Model 0 version of the program for subsequent simulations.

Analysis Results

The resulting values of the more significant SPS performance parameters, as determined in the analysis, are presented in Tables 4 and 5. Table 4 contains values for the third burn as computed in the PAP simulation. Values are presented for three time slices, which were selected to show performance before and after crossover and at both normal and increase PU valve positions. Table 5 contains the flight performance values for the seventh burn from the PAP analysis. The values shown are for two representative time slices following FS-1. In both tables the corresponding preflight predicted values for the same time slice are also shown. All performance values, both predicted and from the PAP analysis, are at the same PU valve position and should be directly comparable.

Figures 2 and 3 show the calculated SPS specific impulse, propellant mixture ratio, and thrust, as functions of time, for the third burn and the seventh burn, respectively. For comparison the figures also contain the predicted performance. As shown, the specific impulse was between 314.0 and 314.2 seconds throughout both burns. Based on the values computed for the two burns analyzed, and the qualitative comparison of the data from all seven burns, it is concluded that the SPS steady-state performance throughout the entire mission was satisfactory. The propellant mixture ratio agreed well with predicted. It should be noted that the predicted performance for this mission incorporated a mixture ratio bias in order to more closely predict the decreased mixture ratio observed on recent flights. A more detailed comparison of the flight performance to the predicted performance is contained in the following section.

The PAP analysis of the third burn determined that the best match to the available data required that the engine fuel hydraulic resistance be adjusted from its acceptance test value. The derived fuel resistance was 820.0 lbf-sec^2/lbm-ft^5, which is approximately 6.7% less than the value determined from engine acceptance test data. Similarly, the fuel resistance derived in the seventh burn analysis was 819.9 lbf^2/lbm-ft^5 which agrees well with the third burn results. A minor adjustment (-1%) was also required in the oxidizer resistance.

Significant biases were found to exist in both interface and both propellant tank pressure measurements. Prior to each burn the measured fuel tank pressure (SP0006 P) consistently indicated 3-4 psi less than the fuel interface pressure (SP0930 P) and the measured oxidizer tank pressure (SP0003 P) read 1-3 psi greater than the oxidizer interface pressure (SP0901 P). During coast, the respective tank and interface pressure should be essentially the same. Interpreting the fuel pressure coast discrepancy as a negative tank pressure bias appeared consistent with the simulation results from both burns, which indicated that the measured fuel tank pressure was biased by approximately -3 psi. The oxidizer pressures, however, were not so easily rectified. The burn simulations indicated that both the oxidizer interface and tank pressure were negatively biased, with the measured oxidizer interface pressure averaging 6.5 psi less than the simulated pressure and the measured oxidizer tank pressure averaging 4 psi less than the simulated pressure. Large (2 to 4 psi) negative oxidizer interface pressure biases under flow conditions have been observed in previous postflight analyses (References 4, 5, 6, and 7) and therefore approximately 3 psi of the 6.5 psi interface pressure bias were expected. The remaining 3.5 psi of the 6.5 psi oxidizer interface pressure bias, and the corresponding -4.0 oxidizer tank pressure bias are somewhat suspect since they are both negative, which indicates the two measurements agree after correcting for a -3 psi flow bias on the interface pressure. However, use of the unbiased tank pressure (and corresponding net interface pressure bias of only -2.5 to -3 psi) resulted in oxidizer flow rates, and thrust levels significantly less than indicated by the PUGS data and the measured acceleration. Furthermore, the resulting oxidizer tank pressure was significantly less (3-4 psi) than the computed fuel tank pressure, which is not expected since both are controlled by the same helium regulator. Therefore, it was concluded that a -6.5 and a -4 psi bias existed in the measured oxidizer interface and tank pressures, respectively.

The analysis verified that the thrust chamber throat area characterization (Figure 1) was relatively accurate, in that no changes were required to achieve a satisfactory data match for either the third or seventh burn. Both the third and seventh burn PAP analysis indicated that the initial estimates of the spacecraft damp weight were essentially correct with no changes being required.

Early analysis results indicated inconsistencies in the amounts of propellants that were reportedly loaded, the amounts indicated by the tank gages and the simulation results. These discrepancies were most apparent after crossover and were therefore associated with the sump tank gages. In general, simulations which best matched the sump tank gages after crossover (near the end) of the third burn required either unreasonably large (approximately 150 pounds of oxidizer and 90 pounds of fuel) reductions in the estimated initial propellant masses onboard at the start of the burn segment, or flowrates (and thrust) which did not agree with the storage tank probes and acceleration data. Since the first two burns were of relatively short duration, the propellant loads onboard at the beginning of the third burn should be known to almost the loading tolerances. Furthermore, when the propellants remaining at the end of the third burn for these simulations were extrapolated to the seventh burn, the same sump tank probes indicated that the extrapolated quantities (and therefore the computed quantities remaining at the end of the third burn) were too low by 100-200 pounds for each propellant. Although the extrapolation from the end of the third burn segment to the start of the seventh burn segment was larger (about 100 seconds of total burn time) than in previous postflight analyses, and therefore the uncertainties in the extrapolations are larger, these discrepancies between burns seemed unreasonable.

The fuel discrepancies were resolved satisfactorily by reducing the fuel onboard at the start of the third burn segment by 34 pounds from the value extrapolated from KSC loading data, and by applying scale factors of 0.992 and 0.994 to the fuel storage and sump tank gages, respectively. With these corrections, the final simulation gave a fuel mass at the end of the third burn segment which, when extrapolated to the seventh burn, was within 80 pounds of the value determined from the final seventh burn simulation. The adjustments required were all within the loading, PUGS, and extrapolation uncertainties.

The oxidizer discrepancies could not be fully resolved. The final third burn simulation assumed the initial oxidizer mass to be as extrapolated from the KSC reported load, and gave an oxidizer mass at the end of the burn segment which, when extrapolated to the seventh burn gave excellent agreement (within 50 pounds of the seventh burn simulation results). However, as shown in Figure 9, the third burn simulation results did not agree with the oxidizer sump tank probe after crossover. The simulation gave approximately 150 pounds more onboard than the sump tank probe indicated at the end of the burn segment. This disagreement could not be simply explained since the same probe did agree well with simulation results for seventh burn, which discounts a constant bias or scale factor error. It is conceivable that a probe bias or scale factor error existed for the third burn, but did not exist during the seventh burn; or that the storage tank did not completely empty at crossover. This second possibility could occur and not be directly detected because the preset -0.4% calibration bias on storage tank probe would make the storage tank probe indicate zero even with 100 pounds of oxidizer in the tank. The large positive slope (Figure 9) on the oxidizer sump tank residual error after crossover could indicate some continued flow into the sump tank for the remainder of the third burn, whereas the simulation assumes the storage tank to be empty at crossover.

In spite of the above third burn oxidizer gaging inconsistencies, the simulation computed consumption (Table 6) for the whole mission agrees quite well with consumption computed from the reported KSC loads and the gaging system readings at shutdown of the seventh burn. Based on the simulation results the total oxidizer and fuel consumed were 23889 pounds and 14932 pounds, respectively. The corresponding values computed from the reported loads and the gage readings (accounting for seventh burn shutdown consumption) were 23900 pounds and 14953 pounds. Based on the computed consumption the overall mission mixture ratio was 1.600, which indicates excellent propellant management. Following the end of the seventh burn the computed usable[1] oxidizer and fuel quantities remaining were 877 pounds and 583 pounds, respectively. Based on the spacecraft mass at the end of the seventh burn, the estimated SPS ΔV capability remaining was approximately 580 ft/sec[2], which exceeds the 500 ft/sec budgeted (Reference 8) for weather avoidance.

(1) Based on unusable quantities of 295.2 pounds and 146.2 pounds for oxidizer and fuel, respectively.

(2) Includes additional allowance of 100 pounds unusable for the $\pm$100 pound PU unbalance meter control ("green") band.

Shown in Figures 4 through 21 are the PAP output plots which present the residuals (differences between the filtered flight data and the program-calculated values) and filtered flight data for the segments of the second and sixth burns analyzed. The figures appear in the following order: vehicle thrust acceleration, oxidizer tank pressure, fuel tank pressure, oxidizer interface pressure, fuel interface pressure, oxidizer sump tank quantity, fuel sump tank quantity, oxidizer and fuel storage tank quantities (second burn only), and chamber pressure for the third and seventh burn, respectively. The values for slopes and intercepts seen in the upper right hand corner of these graphs represent the slopes and intercept on the ordinate of a linear fit of the residual data. It is readily seen that the closer these numbers are to zero, the better the match.

A strong indication of the validity of the PAP simulation can be obtained by comparing the thrust acceleration calculated in the simulation to that derived from the Apollo Command Module Computer (CMC) ΔV data transmitted via measurement CG0001V. This comparison is easily made in terms of the previously mentioned residual slope and intercept data. Figures 3 and 13 show the thrust acceleration during the portions of the burns analyzed, as derived from the CMC data, and the residual between the data and program calculated values. The residual time histories have essentially zero means and little, if any, discernible trend. This indicates that the simulations especially in terms of the computed specific impulse, are relatively valid, although other factors must also be considered in critiquing the simulations.

As observed on previous flights, the measured chamber pressure drifted with burn time during both burns, presumably because of thermal effects on the transducer. Although this drift has been partially modeled from knowledge obtained from past SPS flight analysis results, the existing drift model is, at best, approximate and not sufficient for detailed performance analyses where chamber pressure errors of less than 0.5 psi are significant. Because of the questionable nature of the chamber pressure data, this measurement was considered essentially useless for the detailed analysis, and was not used in the simulations. The residuals plots, Figures 12 and 20 for the chamber pressure during the third and seventh burns are included for information only. Because the chamber pressure could not be utilized, the ability of PAP to distinguish tank and interface pressure measurement errors from errors in the preflight engine model (engine resistances, thrust chamber characteristic velocity, and specific impulse) was somewhat diminished.

Several of the residual plots for the third burn show discontinuities at the times where PU valve movements and propellant crossover occur. These discontinuities are the result of the transients associated with the changes in interface pressure and are not considered significant errors in the match. It should be pointed out that the large number of PU valve movements made during the third burn increases the difficulty of achieving a good simulation and, therefore, the degree of data match achieved is considered quite good.

Comparison With Preflight Performance Prediction

Prior to the Apollo 14 Mission, the expected performance of the SPS was presented in Reference 2. This performance prediction was for the integrated propellant feed/engine system and, wherever possible, utilized data and characteristics for the specific SPS hardware on this flight.

The predicted steady-state thrust, propellant mixture ratio, and specific impulse are shown in Figures 2 and 3 for the third and seventh burns, respectively. Also shown, for comparison, are the corresponding values for the flight as determined from the steady-state analysis. In general, the comparison of the flight performance to the preflight predicted performance is not straightforward because of the difference in the predicted and actual PU valve position history. The slightly different PU valve position history resulted from the mixture ratio, at a given PU valve position, being somewhat greater than predicted. However, valid comparisons can be made at those times where the predicted and actual PU valve positions are the same.

Previous flight results have consistently shown the inflight mixture ratio to be significantly less than expected based on the engine acceptance test data. In order to more closely predict the expected inflight mixture ratio based on past flight experience, the engine fuel hydraulic resistance determined from the acceptance test data was biased by -5.5%, which decreased the mixture ratio obtained with the acceptance test hydraulic resistances by 2.8% at standard inlet conditions. This bias was statistically obtained from the results of the postflight evaluations of Apollo Missions 9, 10, 11, and 12. (Reference 9).

As can be seen in Figures 2 and 3, at common PU valve positions, the flight reconstructed mixture ratio agrees quite well with the predicted mixture ratio. The maximum difference of 0.02 is well within preflight uncertainty (Reference 9) of ±0.047 (3σ). Similarly, the reconstructed thrust and specific impulse (Figures 2 and 3) were within the prediction uncertainties of ±280 pounds (3σ) and ±1.59 seconds (3σ), respectively, at common PU valve positions.

Engine Performance at Standard Inlet Conditions

The expected flight performance of the SPS engine was based on data obtained during the engine and injector acceptance tests. In order to provide a common basis for comparing engine performance, the acceptance test performance is adjusted to standard inlet conditions. This allows actual engine performance variations to be separated from performance variations which are induced by feed-system, pressurization system, and propellant temperature variations.

Based on the steady-state analysis of the third burn, the standard inlet conditions thrust, specific impulse and propellant mixture ratio were 20731 pounds, 313.9 seconds and 1.557, respectively. These values are 0.4% greater, 0.03% greater and 0% different, respectively, than the corresponding values computed from the engine model used in the preflight prediction.

The seventh burn analysis yielded standard inlet conditions thrust, specific impulse and propellant mixture ratio of 20741 pounds, 314.1 seconds and 1.557 units, respectively. These values are 0.4% greater, 0.1% greater and 0% different, respectively, than the corresponding values computed from the preflight engine model.

The standard inlet conditions performance values for the two burns agree well with each other, with the thrust, specific impulse, and propellant mixture values being only 10 pounds, 0.2 seconds, and 0.0 units different, respectively. The average standard inlet conditions thrust, specific impulse and propellant mixture ratio for the two burns were 20736 pounds, 314.0 seconds and 1.557 units, respectively. These values are 0.4% greater, .1% greater, and 0% different, respectively, than the corresponding values computed from the preflight engine model.

As previously discussed, the engine fuel resistance used in the preflight prediction was adjusted from its acceptance test value in an attempt to improve the mixture ratio prediction. If the average standard inlet conditions thrust, specific impulse and mixture ratio from the flight are compared to their corresponding values computed from an engine model based on the unadjusted acceptance test resistances the flight values are found to be 1% greater, 0.1% greater and 2.9% less, respectively, than the values from the unadjusted model.

The standard inlet conditions performance values reported herein were calculated for the following conditions.

STANDARD INLET CONDITIONS

Oxidizer interface pressure, psia	162
Fuel interface pressure, psia	169
Oxidizer interface temperature, °F	70
Fuel interface temperature, °F	70
Oxidizer density, lbm/ft^3	90.15
Fuel density, lbm/ft^3	56.31
Thrust acceleration, lbf/lbm	1.0
Throat area (initial value), in^2	121.700

Of primary concern in the flight analysis of all Block II engines is the verification of the present methods of extrapolating the specific impulse for the actual flight environment from data obtained during ground acceptance tests at sea level conditions. Since the SPS engine is not altitude tested during the acceptance tests, the expected specific impulse is calculated from the data obtained from the injector sea level acceptance tests using conversion factors determined from Arnold Engineering Developing Center (AEDC) simulated altitude qualification testing. As previously discussed, the average standard inlet conditions specific impulse determined from analyses of the third and seventh burns was 314.0 seconds. The predicted specific impulse at standard inlet conditions, as extrapolated from the ground test data was 313.8 seconds. The expected tolerance associated with the predicted standard inlet condition value of 313.8 seconds (Reference 2) was ±1.593 seconds (3-sigma). The flight value was well within this tolerance. Therefore, it is concluded that the present methods of extrapolating the expected flight specific impulse from the ground test data were satisfactory for this flight, and there is no evidence to warrant changing the methods for future flights. The validity of this conclusion should be continually verified on each subsequent flight.

5. PUGS EVALUATION AND PROPELLANT LOADING

Propellant Loading

The oxidizer tanks were loaded to CM display readout of 100.85% at a tank pressure of 111 psia and an oxidizer temperature of 67.7°F. The fuel tanks were loaded at 110 psia and 69.5°F to a display readout of 100.8%. The SPS propellant loads calculated from these data, and propellant sample density data, are shown in Table 6. As planned, the oxidizer storage tank primary gage was zero adjusted with an approximate -0.4% bias. This zero adjustment bias was incorporated for Apollo 10 and subs to prevent erroneous storage tank readings after crossover as experienced during the Apollo 9 Mission (Reference 3). The zero adjustment bias causes a small, but known, time varying error (a -0.4% bias and a +0.8% scale factor) in the readings from the storage tank primary gage prior to crossover.

PUGS Operation in Flight

The propellant utilization gaging system (PUGS) operated satisfactorily throughout the mission. The PUGS mode selection switch was set in the normal position for all SPS burns, therefore, only the primary system data were available. The propellant utilization (PU) valve was in the increase position at launch and for the first two burns.

The PU valve was in the increase position for the start of the third burn. Approximately 104 seconds after ignition of the third burn, the PU valve was moved to the normal position. At approximately 195 seconds following ignition the PU valve was returned to the increase position. The valve was moved to the normal position again at about 285 seconds following ignition, and back to increase again at about 320 seconds after ignition. The valve was left in the increase position for the remainder of the third burn, all the fourth, fifth and sixth burns, and for approximately the first 18 seconds of the seventh, at which time it was moved to the normal position. The PU valve was left in the normal position for the remainder of the seventh burn (approximately 131 seconds).

Figure 22 shows the indicated propellant unbalance history for the third and seventh burns; as computed from the raw T/M PUGS data. The indicated unbalance history should reflect the CM display unbalance history, within the T/M accuracy, and, in fact, agrees well with the crew reported unbalance readings of 40 pounds, increase and 40 pounds, decrease at the end of the third and seventh burns, respectively. The PU valve position history, as shown in Figure 22, was, as best could be determined from the T/M data, consistent with the indicated unbalance and the expected crew positioning logic (Reference 2). The expected unbalance and associated PU valve position history which are also shown in Figure 22, are seen to compare favorably with the actual. As expected, based on past flights, the indicated unbalance following the start of the third burn showed decrease readings. The initial decrease readings are caused by three factors: 1) the previously mentioned -0.4% calibration bias on the oxidizer storage tank probe, 2) ungageable oxidizer (approximately 100 pounds) above the top of the oxidizer sump tank probe prior to crossover due to propellant transfer resulting from helium absorption, and 3) the tendency of the fuel probes to read erroneously high for about 30-40 seconds following ignition on low acceleration burns due to capillary action in the probe stillwells. After propellant crossover, at about 240 seconds of the third burn, the unbalance is seen to take a step increase to approximately zero as the effects of the two known oxidizer sump tank gaging errors (the -0.4% bias and the oxidizer above the probe) are eliminated. Throughout the balance of the third burn and during the seventh burn the unbalance was satisfactorily controlled within the ±100 lbm unbalance "green band" by use of the PU valve. At the end of the seventh burn the unbalance was reported by the crew as only 40 pounds, decrease.

6. PRESSURIZATION SYSTEM EVALUATION

Operation of the helium pressurization system was satisfactory without any indication of leakage. The helium supply pressure and the propellant ullage pressures indicated nominal helium usage for the seven SPS burns.

The propellant tanks were pressurized several days prior to launch, and at liftoff the measured tank pressures were approximately 180 psia for oxidizer and 179 psia for fuel.

During the launch phase and coast period to the first SPS burn, the measured oxidizer and fuel tank pressures decayed, as expected, to approximately 166 psia and 173 psia, respectively, due primarily to helium absorption into the propellants.

This mission was the first to utilize the SPS engine to perform the DOI maneuver. The very precise ΔV requirements for this burn made any over or underburn highly undesirable, and, therefore, a crew timed backup cutoff was implemented. Because of the critical nature of this burn an analysis was performed (Reference 2) to determine the best estimate for the propellant tank pressure rises from the end of the LOI burn to the start of the DOI burn since thrust level and therefore burn time, are dependent on the initial tank pressures. Such propellant tank pressure rises have been experienced on past Apollo flights and are attributed to propellant vapor resaturation and temperature recovery of the ullage which occur following a long burn in which there is a significant percentage increase in ullage volume. The predicted pressure rises were 8.5 psia for the oxidizer tank and 5 psia for the fuel tank, and the pressure rises experienced on Apollo 14 were 9 psia for the oxidizer tank and 3.5 psia for the fuel tank.

The GN_2 actuation system pressures indicated satisfactory usage. At launch the storage pressures for GN_2 Systems A and B were both approximately

2375 psia. Following the seventh and final SPS burn the T/M data indicated that the System A pressure was 2040 psia and that the System B pressure was 2165 psia. System A was utilized on all seven SPS burns for an indicated average pressure decrease of approximately 48 psia per burn. System B was utilized on five burns for an indicated average pressure decrease of 42 psia per burn.

7. ENGINE TRANSIENT ANALYSIS

A summary of the start and shutdown transient performance data for the seven SPS firings is presented in Table 7. The start impulse for the fourth, fifth, and sixth burns exceeded the upper specification limit of 700 lbf-sec by 67 lbf-sec, 6 lbf-sec, and 23 lbf-sec, respectively. The difference between the first and third burns, and the fourth, fifth, and sixth burns exceeded the start impulse run-to-run specification limits of ±200 lbf-sec. The start time (ignition to 90 percent of steady-state thrust) for each burn were all within the specification limits. The computed shutdown impulse for the third burn was less than the lower specification limit of 10,000 lbf-sec, and the variability between burns was within the ±500 lbf-sec specification limits, with the exception of the third and seventh burns. The shutdown time (cutoff to 10 percent of steady-state thrust) for each burn were all within specification limits. The out of specification values are not considered significant because: 1) they are relatively minor deviations; 2) the errors involved in computing impulse from chamber pressure data are considered large relative to the deviations; and 3) the values are consistent with past flight values.

The second SPS burn was of much shorter duration than would normally be performed. The CSM RCS would normally be used to perform such a burn; however, due to the CSM-LM docking difficulties the SPS engine was used for this burn in order to conserve RCS propellants. The total vacuum impulse for this burn was very close to the value reported in the Spacecraft Opertional Data Book (Reference 2).

The engine was started in the single bore mode (engine valve bank A) on all maneuvers. The chamber pressure overshoot values are contained in

Table 7 and except for the first and fourth burns, did not exceed the specified maximum of 120 percent. The chamber pressure overshoot on the first burn was 126 percent and the overshoot on the fourth burn was 127 percent. The second and fifth burns were conducted completely in the single bore mode. For the remaining burns the other bore (engine valve bank) was opened 1 to 4 seconds after ignition.

8.0 REFERENCES

1. NASA Report MSC-04112, "Apollo 14 Mission Report," April 1971.

2. Spacecraft Operational Data Book, SNA-8-D-027, Vol. I, Part 1, Amendment 55, 21 December 1970.

3. TRW Letter 71.6522.4-8, "Apollo 14 Postflight Mass Properties," G. L. Christen to Russell S. Morton, 24 March 1971.

4. TRW Technical Report 11176-H311-RO-00, "Apollo 9 CSM 104 Service Propulsion System Final Flight Evaluation," 4 August 1969.

5. TRW Technical Report 11176-H526-RO-00, "Apollo 10 CSM 106 Service Propulsion System Final Flight Evaluation," 31 March 1970.

6. TRW Technical Report 17618-H019-RO-00, "Apollo 11 CSM 107 Service Propulsion System Propulsion System Final Flight Evaluation," Awaiting Publication.

7. TRW Technical Report 17618-H058-RO-00, "Apollo 12 CSM 108 Service Propulsion System Final Flight Evaluation," November 1970.

8. MSC Internal Note 71-FM-36, "The Consumables Analysis for the Apollo 14 (Mission H-3) Spacecraft Operational Trajectory," 27 January 1971.

9. TRW Technical Report 17618-H129-RO-00, "Apollo Primary Propulsion System Engineering Mathematical Models," March 1971.

TABLE 1

SPS DUTY CYCLE

Maneuver	FS1 (A.E.T.)	FS2 (A.E.T.)	Burn Duration (sec)	Velocity Change (ft/sec)[3]
MCC-2 (Hybrid)	30:36:07.91[2]	30:36:18.10[1]	10.19	71.1
MCC-4	76:58:11.08[1]	76:58:12.63[1]	0.65	3.5
LOI	81:56:40.70[2]	82:02:51.54[1]	370.84	3022.4
DOI	86:10:52.97[2]	86:11:13.78[1]	20.81	205.7
CIRC	105:11:46.11[1]	105:11:50.13[1]	4.02	77.2
LOPC	117:29:33.17[2]	117:29:51.67[2]	18.50	370.5
TEI	148:36:02.30[1]	148:38:31.58[1]	149.28	3460.6
			574.29	7211.0

SOURCE:

(1) Command Module Computer Downlink Data - CG001V

(2) SPS Solenoid Drivers - CH3604X, 5X

(3) Reference 1

TABLE 2

PREDICTED CSM 110 SPS ENGINE AND FEED SYSTEM CHARACTERISTICS

Engine No.	63
Injector No.	121
Chamber No.	343
Initial Chamber Throat Area (in^2)	121.6614

Engine and System Fluid Resistances ($lbf\text{-}sec^2/lbm\text{-}ft^5$)

	Based on Acceptance Test	Adjusted
Fuel Engine Feedline	878.8	830.2
Oxidizer Engine Feedline	488.1	
Fuel System Feedline	36.08	
Oxidizer System Feedline		
PU Valve in Pri-normal position	97.72	
PU Valve in Pri-increase	48.49	
PU Valve in Pri-decrease Position	173.11	

Characterization Equation for C*:

$$C^* = C^*_{S.C.} + 870.5\,(MR - 1.6) - 273.83\,(MR^2 - 2.56) - 0.31878\,(P_C - 99)$$
$$+12.953\,(TP - 70) - 0.07414\,(TP^2 - 4900) - 5.466\,(MR \cdot TP - 112)$$
$$+ 0.03119\,(MR \cdot TP^2 - 7840\cdot);$$

where $C^*_{S.C.}$ (Engine No. 63) = 5970 ft/sec

Characterization Equation for I_{SP}:

$$I_{SP} = I_{SP_{vac}} - 96.954\,(1.6 - MR) - 0.0487\,(99 - P_C) - 0.06276\,(70 - TP)$$
$$+ 30.409\,(2.56 - MR^2) + 0.0004483\,(4900 - TP^2);$$

where $I_{SP_{vac}}$ (Engine No. 63) = 313.8 lbf-sec/lbm

TABLE 3

FLIGHT DATA USED IN STEADY STATE ANALYSIS

Measurement Number	Description	Range	Sample Rate Samples/Sec
SP0930 P	Pressure, Engine Fuel Interface	0 to 300 psia	10
SP0931 P	Pressure, Engine Oxidizer Interface	0 to 300 psia	10
SP0661 P	Pressure, Engine Chamber	0 to 150 psia	100
SP0003 P	Pressure, Oxidizer Tanks	0 to 250 psia	10
SP0006 P	Pressure, Fuel Tanks	0 to 250 psia	10
SP0048 T	Temperature, Engine Fuel Feed Line	0 to 200 °F	1
SP0049 T	Temperature, Engine Oxidizer Feed Line	0 to 200 °F	1
SP0054 T	Temperature, 1 Oxidizer Distribution Line	0 to 200 °F	1
SP0057 T	Temperature, 1 Fuel Distribution Line	0 to 200 °F	1
SP0655 Q	Quantity, Oxidizer Tank 1 Primary - Total Auxiliary	0 to 50%	1
SP0656 Q	Quantity, Oxidizer Tank 2	0 to 60%	1
SP0657 Q	Quantity, Fuel Tank 1 Primary - Total Auxiliary	0 to 50%	1
SP0658 Q	Quantity, Fuel Tank 2	0 to 60%	1
CG0001 V	Computer Digital Data	40 Bits	1/2

TABLE 4

SERVICE PROPULSION SYSTEM STEADY-STATE PERFORMANCE
THIRD SPS BURN

PARAMETER	INSTRUMENTED								
	Before Crossover						After Crossover		
	FS-1 + 50 sec.			FS-1 + 190 sec.			FS-1 + 345 sec.		
	Predicted	PAP	Measured	Predicted	PAP	Measured	Predicted	PAP	Measured
PU Valve Position	Increase	Increase	Increase	Normal	Normal	Normal	Increase	Increase	Increase
Oxidizer Tank Pressure, psia	175	177	174	176	178	174	175	178	175
Fuel Tank Pressure, psia	176	175	172	176	175	173	176	177	174
Ox Interface Pressure, psia	167	169	162	162	163	157	170	173	166
Fuel Interface Pressure, psia	172	171	170	171	171	170	174	175	174
Engine Chamber Pressure, psia	103	103	103	102	102	103	104	105	106
DERIVED									
Oxidizer Flowrate, lbm/sec	41.4	41.8	--	40.0	40.5	--	41.9	42.5	--
Fuel Flowrate, lbm/sec	25.9	25.8	--	26.1	26.1	--	26.0	26.2	--
Propellant Mixture Ratio	1.598	1.619	--	1.533	1.553	--	1.612	1.622	--
Vacuum Specific Impulse, sec	314.0	314.1	--	313.8	314.0	--	314.1	314.2	--
Vacuum Thrust, lbf	21132	21242	--	20742	20891	--	21327	21580	--

Notes:

(1) Predicted values from Reference 2.
(2) Calculated values from Propulsion Analysis Program
(3) Measured data are as recorded and are not corrected for biases and errors in text.

TABLE 5

SERVICE PROPULSION SYSTEM STEADY-STATE PERFORMANCE
SEVENTH SPS BURN

PARAMETER	INSTRUMENTED					
	FS-1 + 50 Sec.			FS-1 + 110 Sec.		
	Predicted	PAP	Measured	Predicted	PAP	Measured
PU Valve Position	Normal	Normal	Normal	Normal	Normal	Normal
Oxidizer Tank Pressure, psia	177	177	173	177	177	174
Fuel Tank Pressure, psia	177	175	173	176	175	173
Oxidizer Interface Pressure, psia	167	166	160	166	166	160
Fuel Interface Pressure, psia	175	173	172	174	173	172
Engine Chamber Pressure, psia	103	103	103	103	103	104
DERIVED						
Oxidizer Flowrate, lbm/sec	40.9	40.9	--	40.6	40.8	--
Fuel Flowrate, lbm/sec	26.4	26.2	--	26.3	26.2	--
Propellant Mixture Ratio	1.549	1.559	--	1.544	1.558	--
Vacuum Specific Impulse, sec	313.9	314.2	--	313.9	314.2	--
Vacuum Thrust, lbf	21125	21085	--	21029	21046	--

Notes:

(1) Predicted values from Reference 2
(2) Calculated values from Propulsion Analysis Program
(3) Measured data are as recorded and are not corrected for biases and errors discussed in text.

TABLE 6

SPS PROPELLANT DATA

Propellant	Total Mass Loaded (lbm) Computed From Loading Data	Based on Analysis
Oxidizer	25061.0	25061
Fuel	15695.2	15661
TOTAL	40756.2	40722

Propellant	Propellant Consumption (lbm) Computed From Loading Data and PUGS	Analysis Results
Oxidizer	23900	23889
Fuel	14953	14932
TOTAL	38853	38821

Propellant	Propellant Residuals (lbm) Computed From PUGS	Analysis Results
Usable Oxidizer	866	877
Usable Fuel	596	583
TOTAL	1462	1460

TABLE 7

ENGINE TRANSIENT DATA

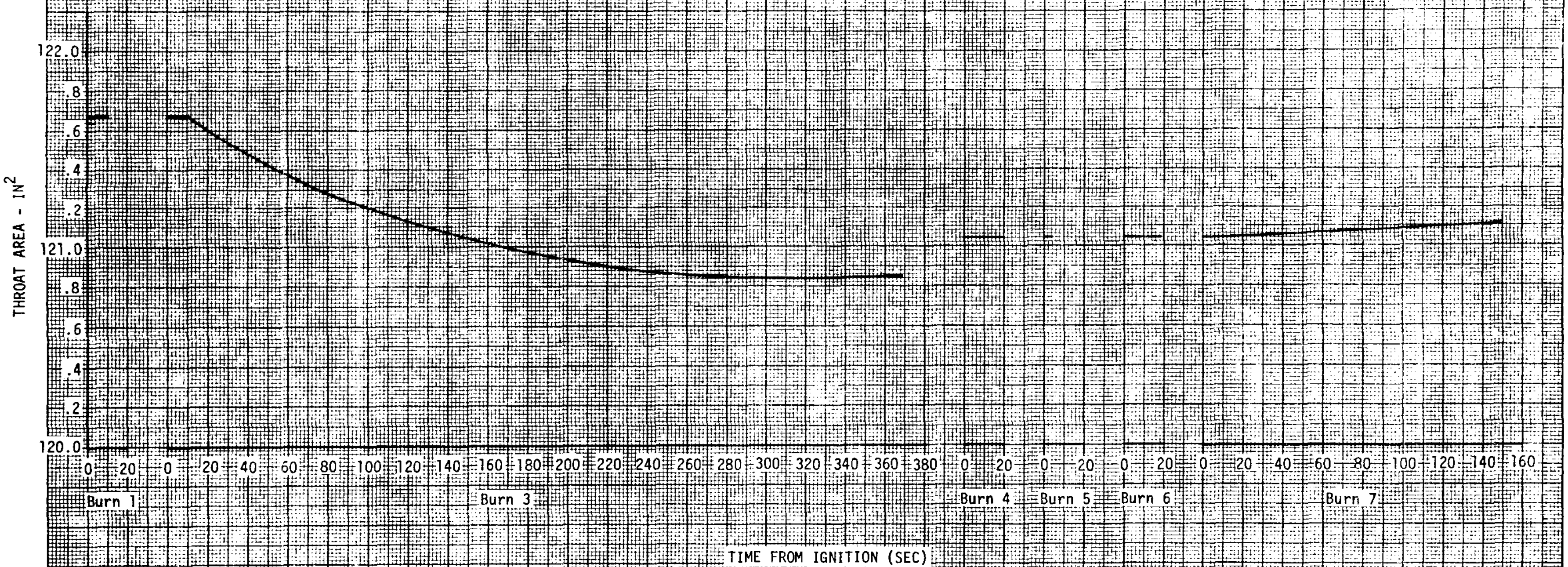
PARAMETER	SPECIFICATION VALUE		APOLLO 14 SPS MANEUVERS						
	Single Bore	Dual Bore	1st	2nd	3rd	4th	5th	6th	7th
Total Vacuum Impulse (Ignition to 90% Steady-State Thrust), lbf-sec	450 ±250 (a) ±200 (b)	568	416		496	767	706	723	592
Time (Ignition to 90% Steady-State Thrust), sec	0.675 ±0.100		.635		.658	.632	.632	.639	.621
Chamber Pressure Overshoot, Percent	120		126	110	118	127	119	115	120
Total Vacuum Impulse (cutoff to 0% steady-state thrust), lbf-sec	12,500 ± 2,500 (a) ± 500 (b)	13,500 ± 2,500 (a) ± 500 (b)	13,090		9,768	12,268	12,250	12,778	14,447
Time (Cutoff to 10% Steady-State Thrust), sec	1.075	1.075	1.030		1.054	1.001	.991	1.049	1.065
Minimum Impulse Burn, Total Vacuum Impulse, lbf-sec	11,700 (c)			12,951					
Time (FS1 to FS2), sec				0.65					

(a) Engine-to-Engine Tolerance
(b) Run-to-Run Tolerance
(c) Reference 2

Note: Maneuver 2 and 5 had single bore starts and shutdowns.
Remaining maneuvers had single bore starts and dual bore shutdowns.

FIGURE 1

SPS CHAMBER THROAT AREA



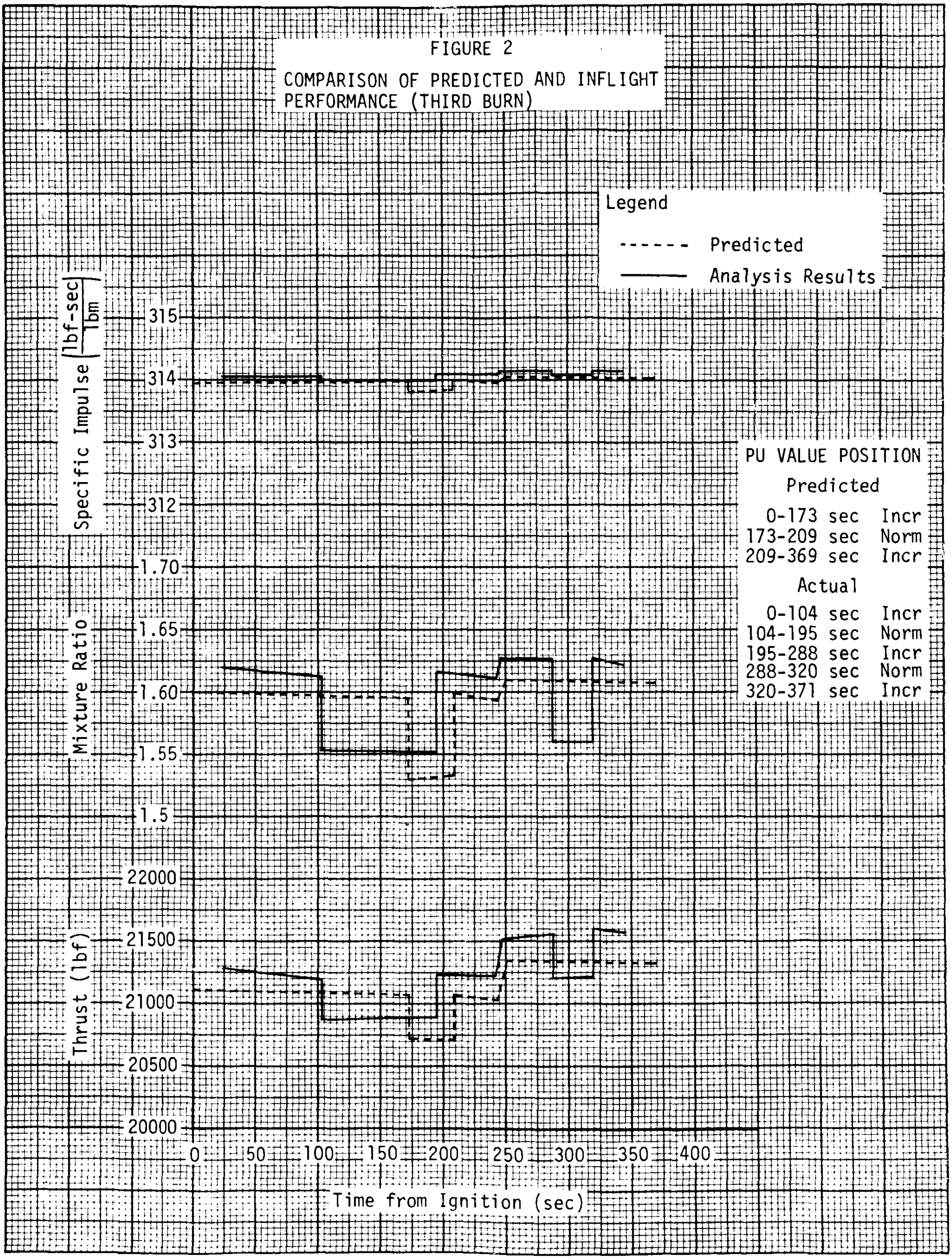


FIGURE 2

COMPARISON OF PREDICTED AND INFLIGHT PERFORMANCE (THIRD BURN)

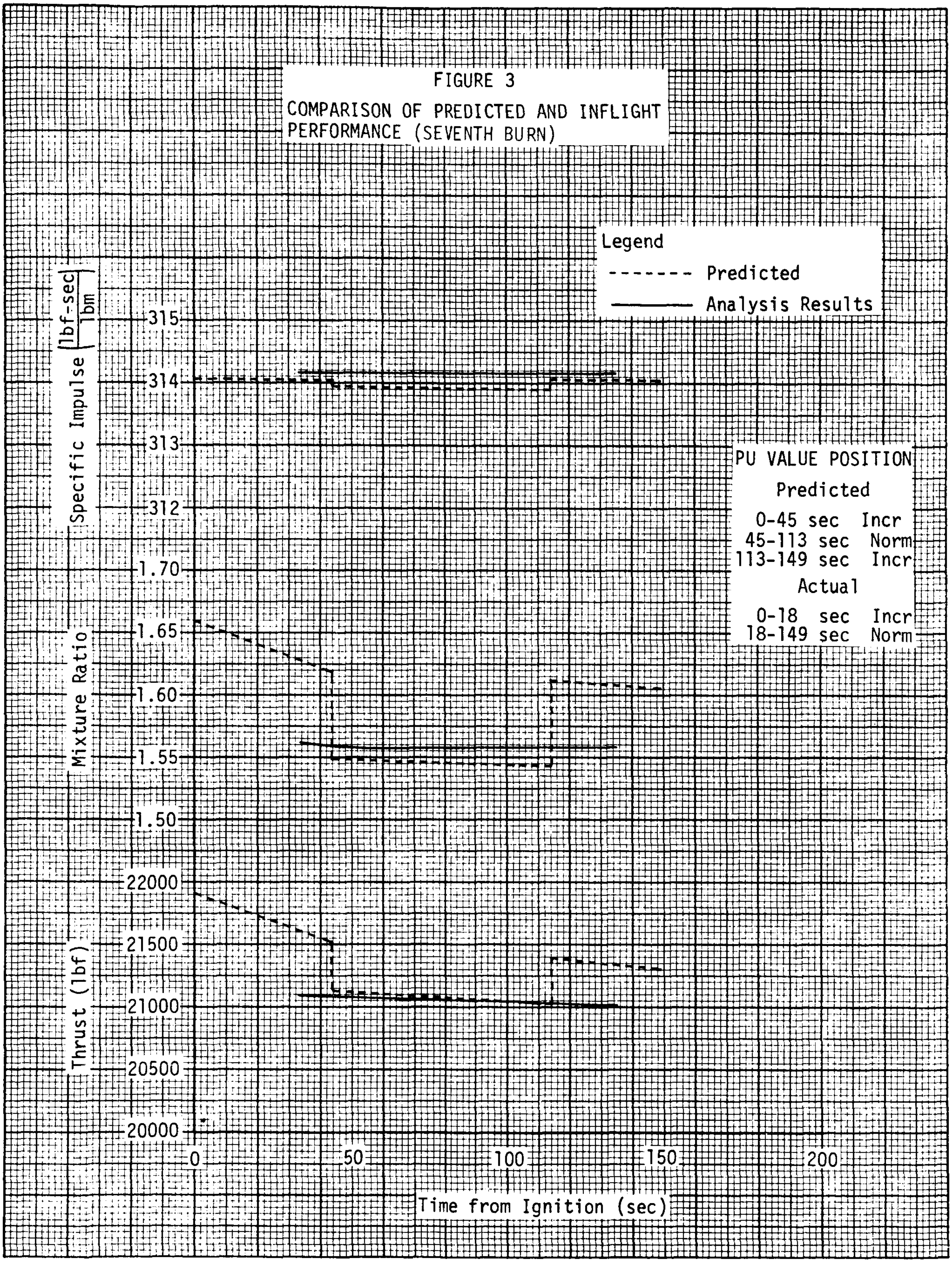


FIGURE 3

COMPARISON OF PREDICTED AND INFLIGHT PERFORMANCE (SEVENTH BURN)

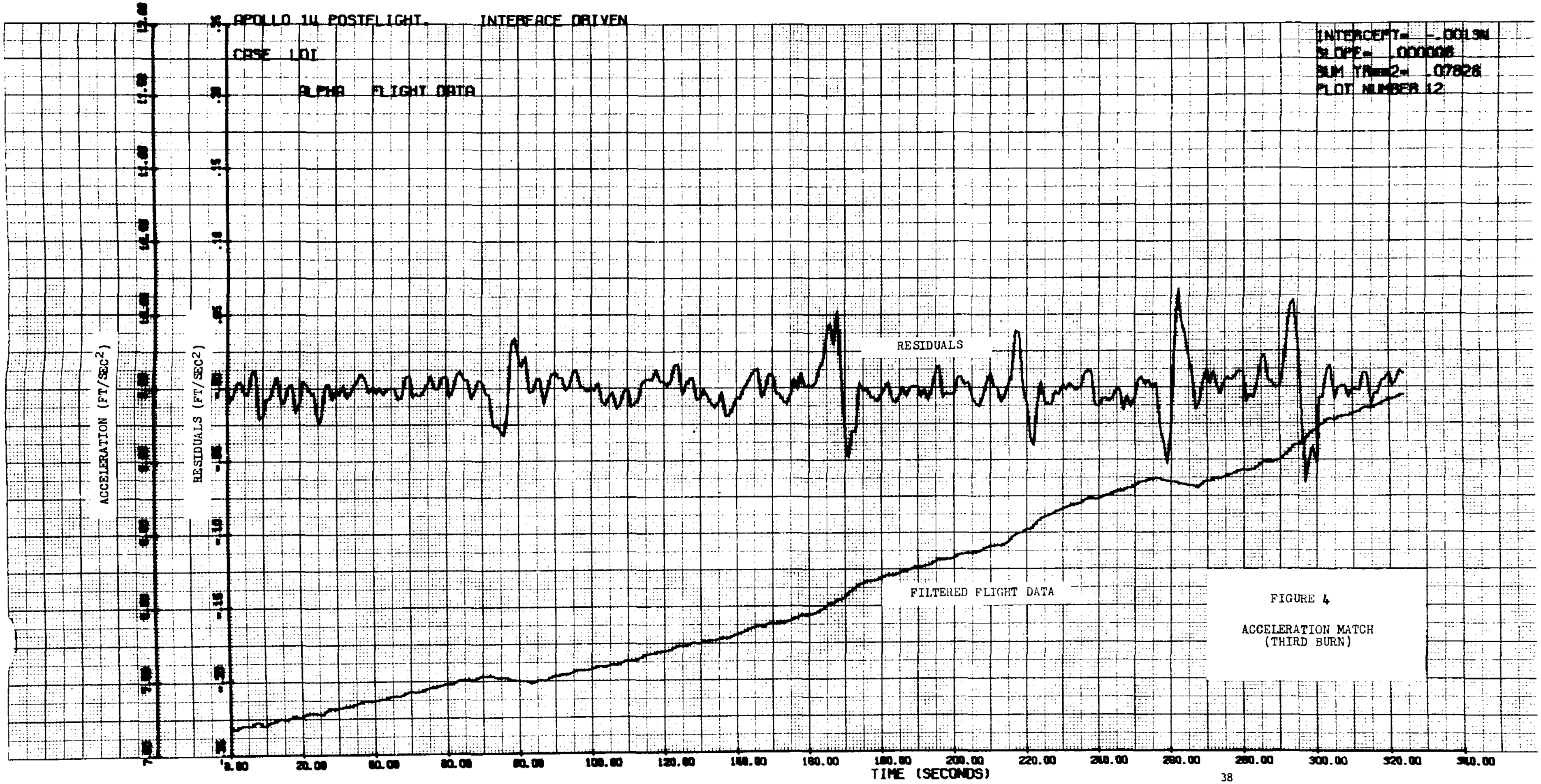


FIGURE 4

ACCELERATION MATCH
(THIRD BURN)

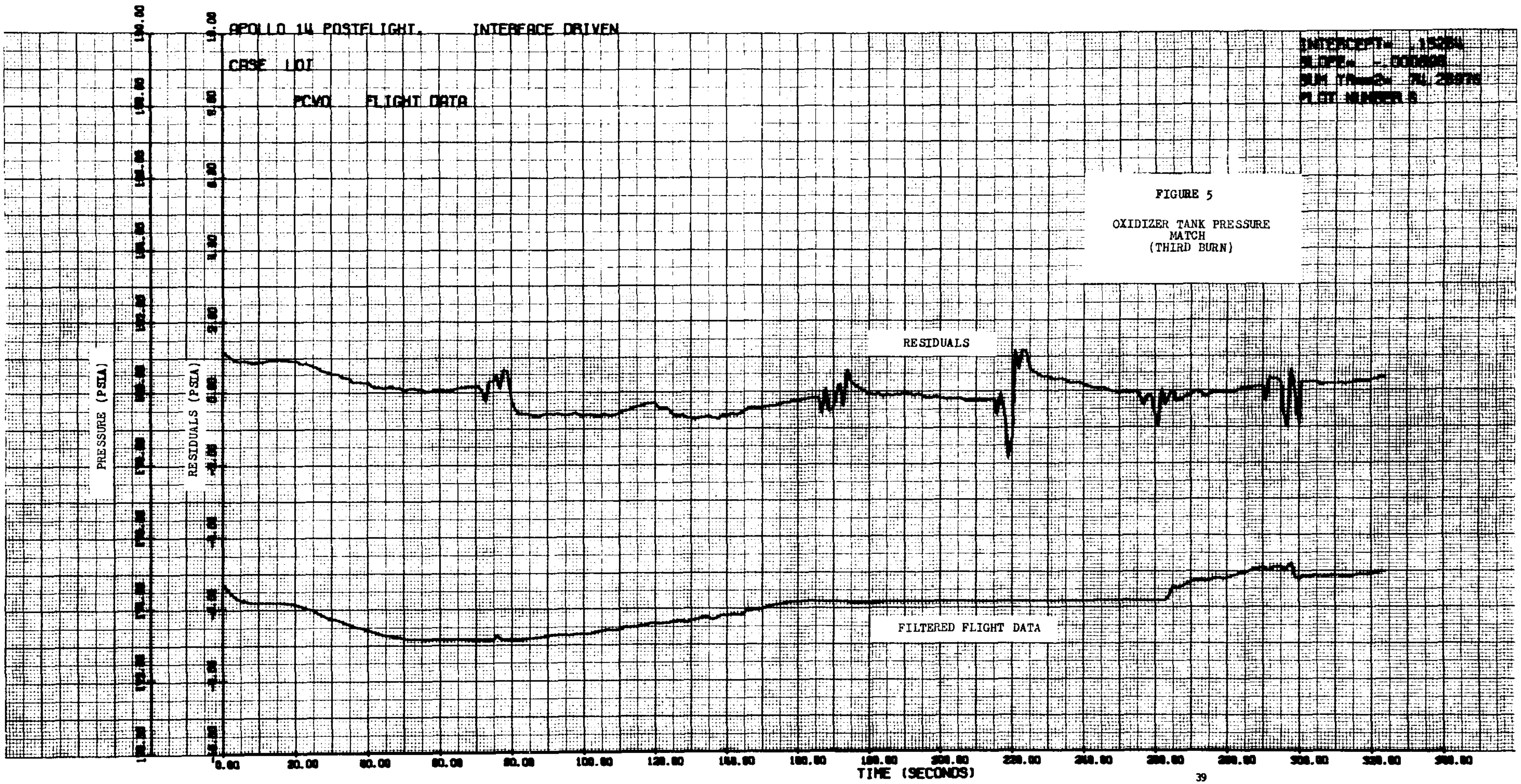


FIGURE 5

OXIDIZER TANK PRESSURE MATCH (THIRD BURN)

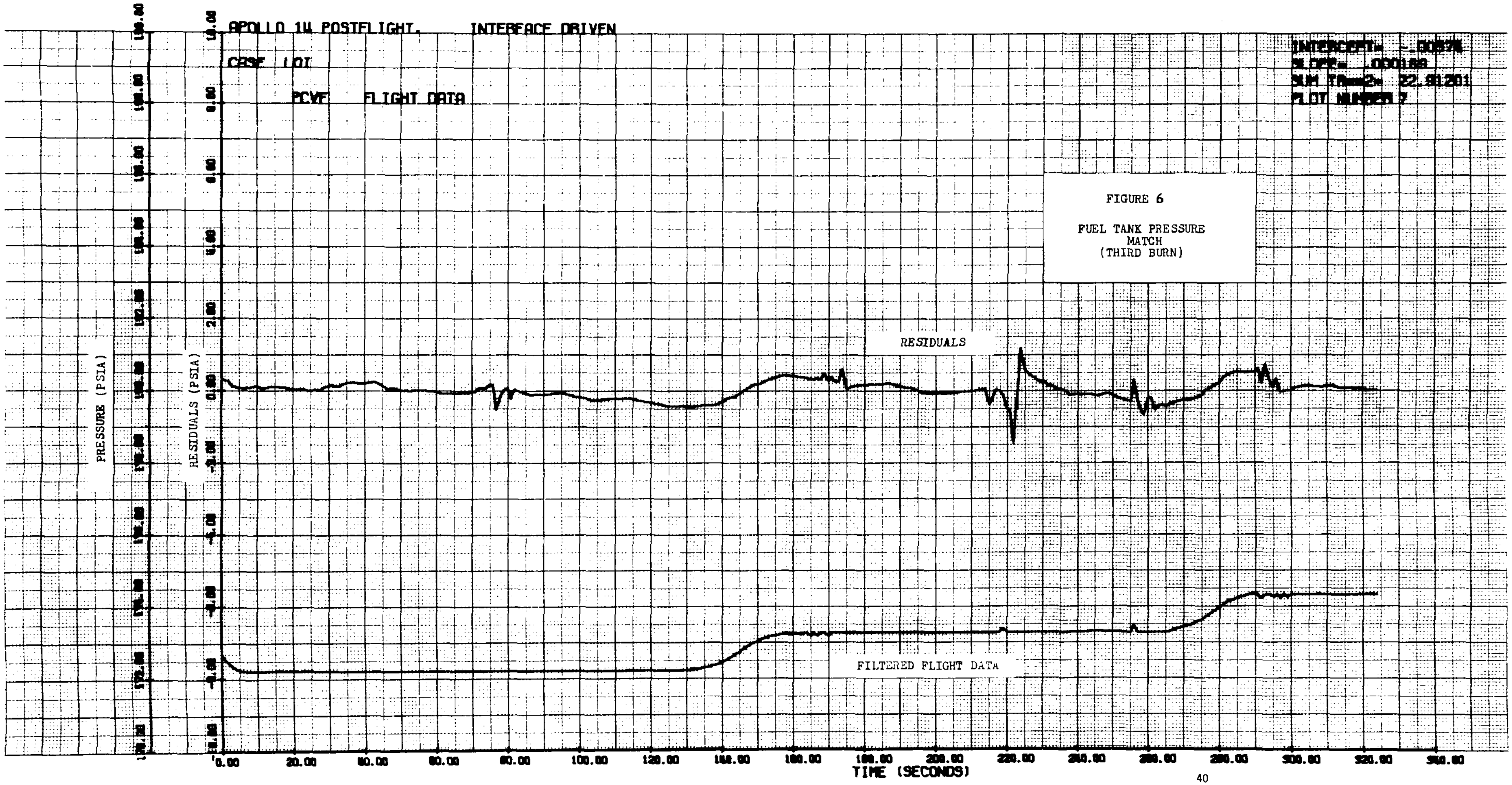


FIGURE 6

FUEL TANK PRESSURE MATCH (THIRD BURN)

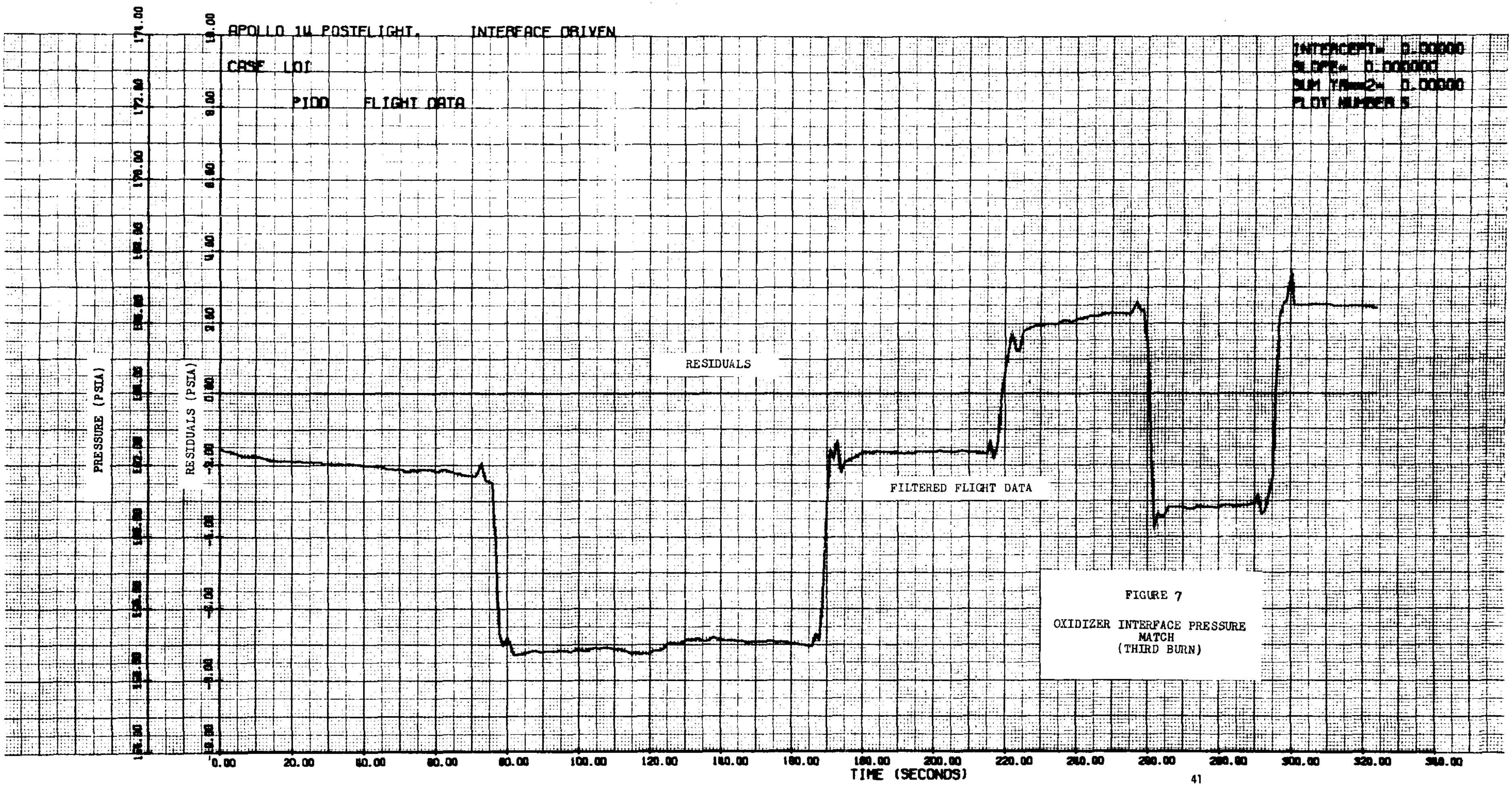


FIGURE 7

OXIDIZER INTERFACE PRESSURE MATCH (THIRD BURN)

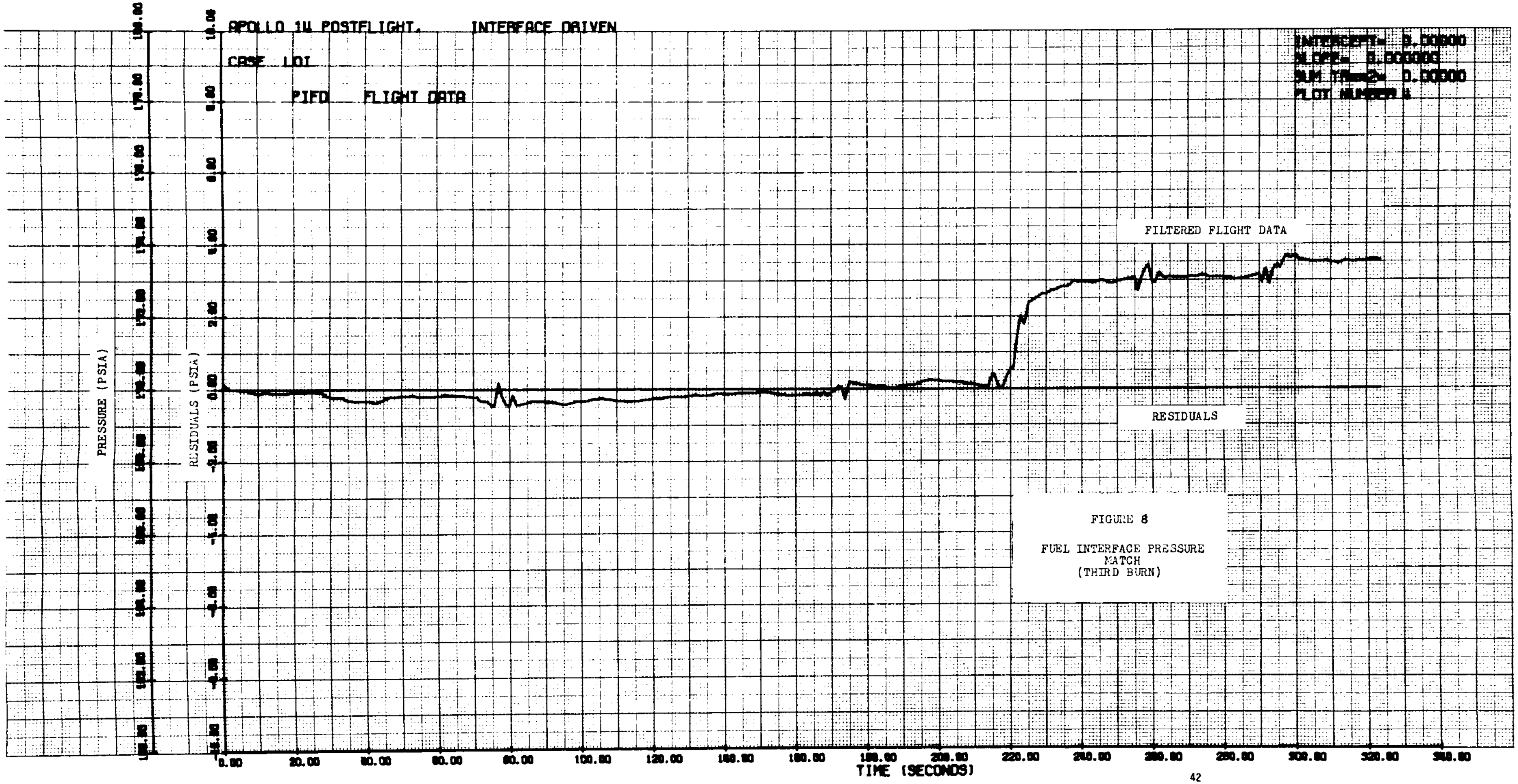


FIGURE 8

FUEL INTERFACE PRESSURE MATCH (THIRD BURN)

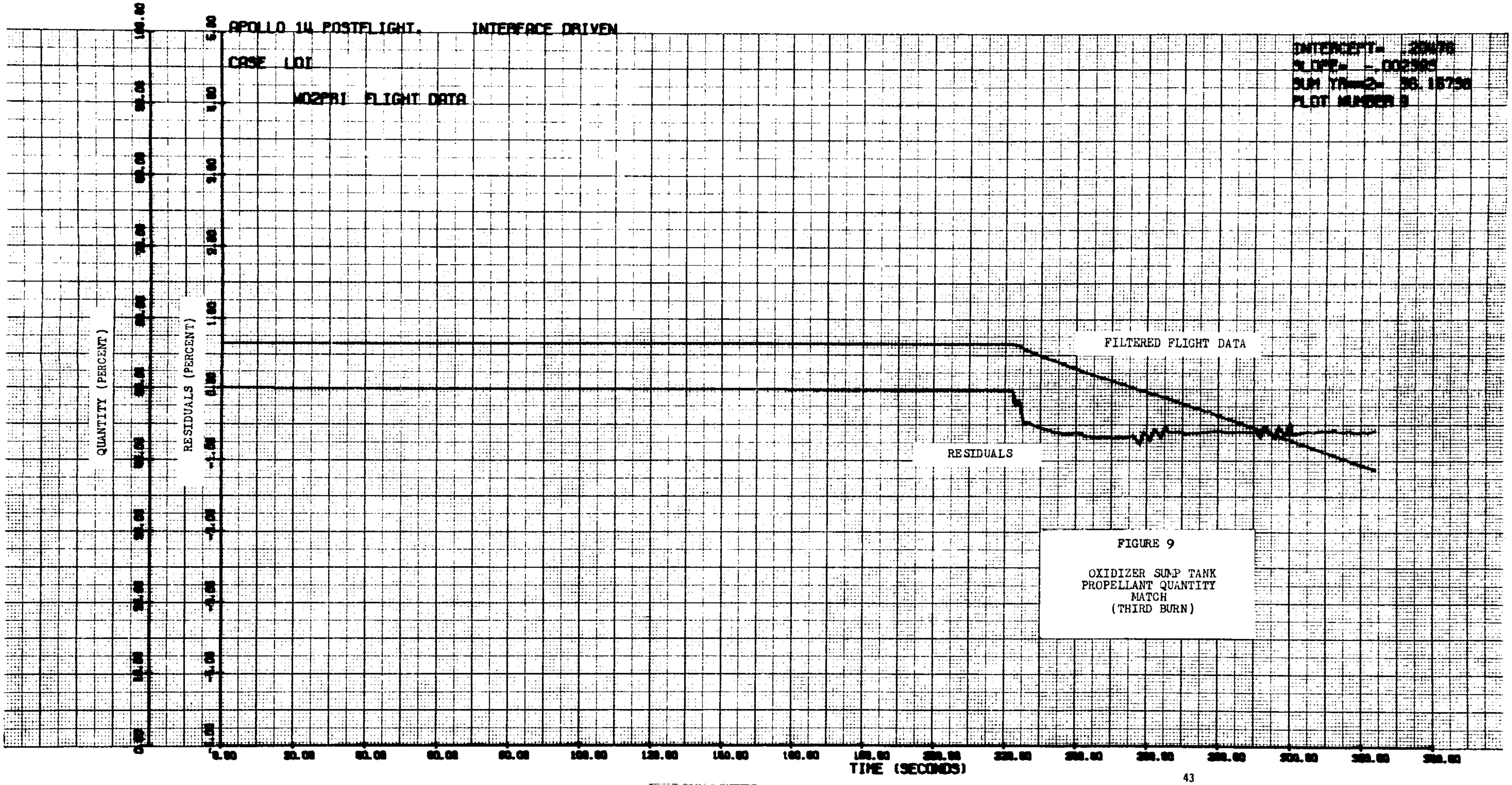


FIGURE 9

OXIDIZER SUMP TANK PROPELLANT QUANTITY MATCH (THIRD BURN)

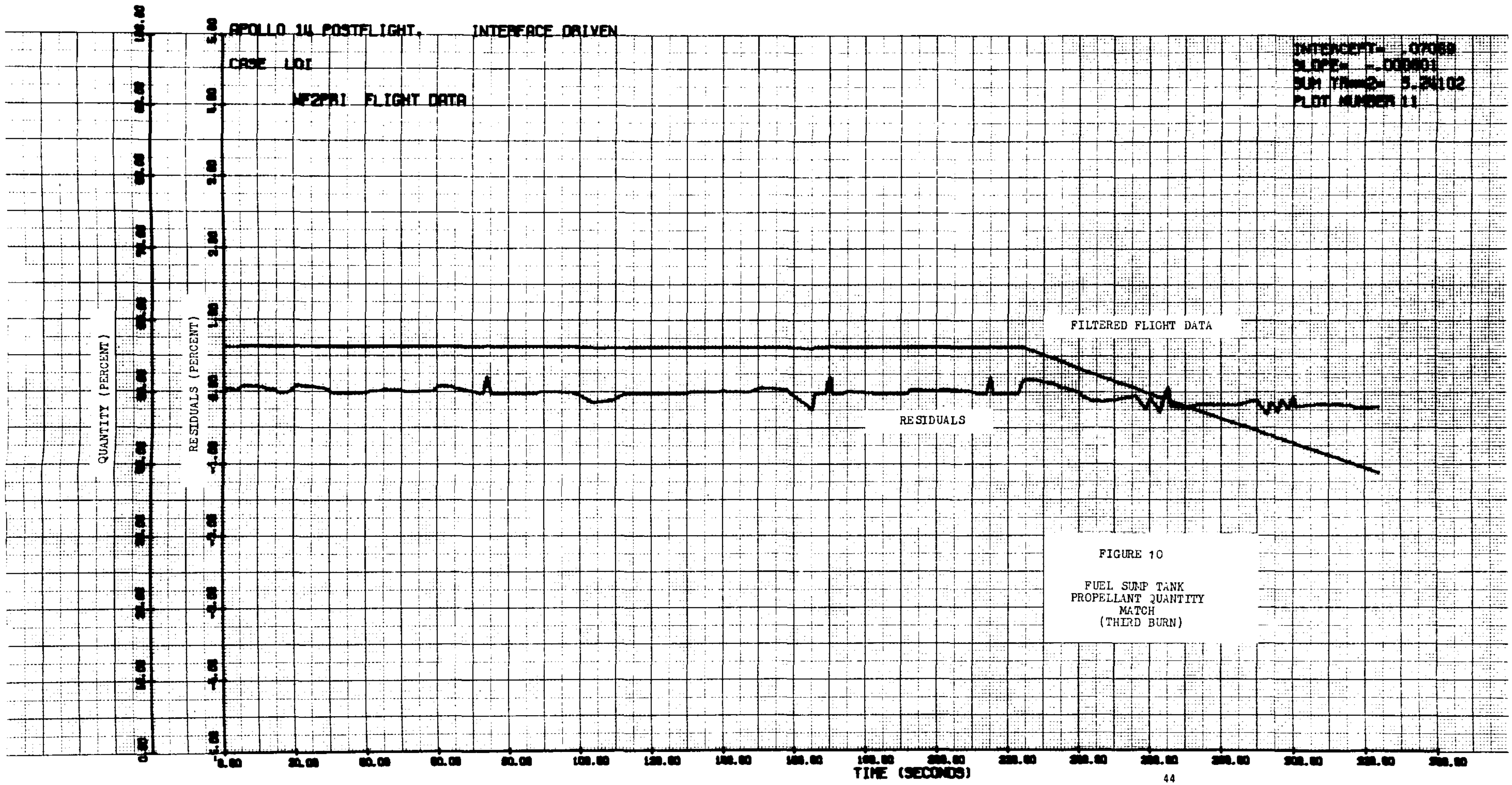


FIGURE 10

FUEL SUMP TANK
PROPELLANT QUANTITY
MATCH
(THIRD BURN)

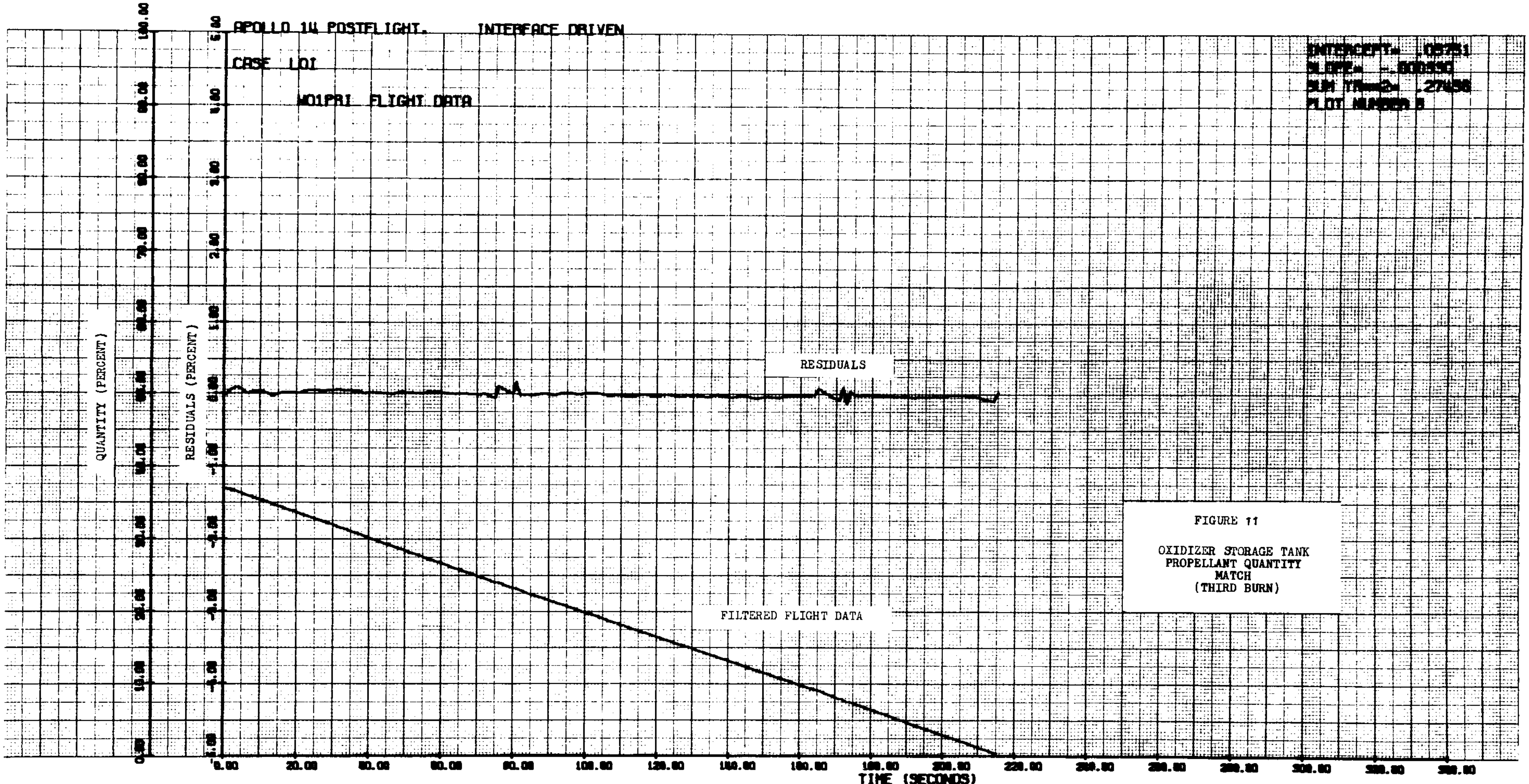


FIGURE 11

OXIDIZER STORAGE TANK
PROPELLANT QUANTITY
MATCH
(THIRD BURN)

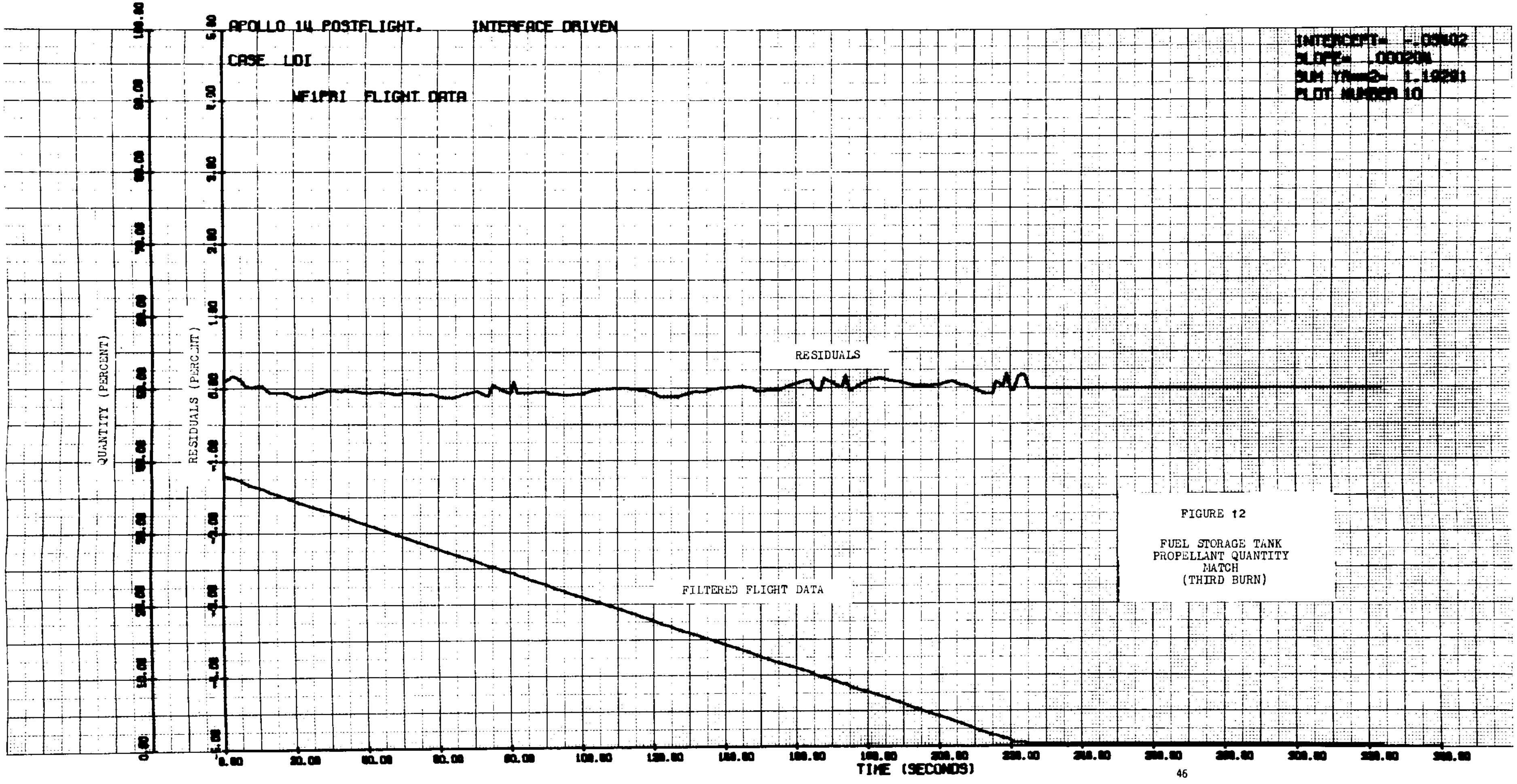


FIGURE 12

FUEL STORAGE TANK
PROPELLANT QUANTITY
MATCH
(THIRD BURN)

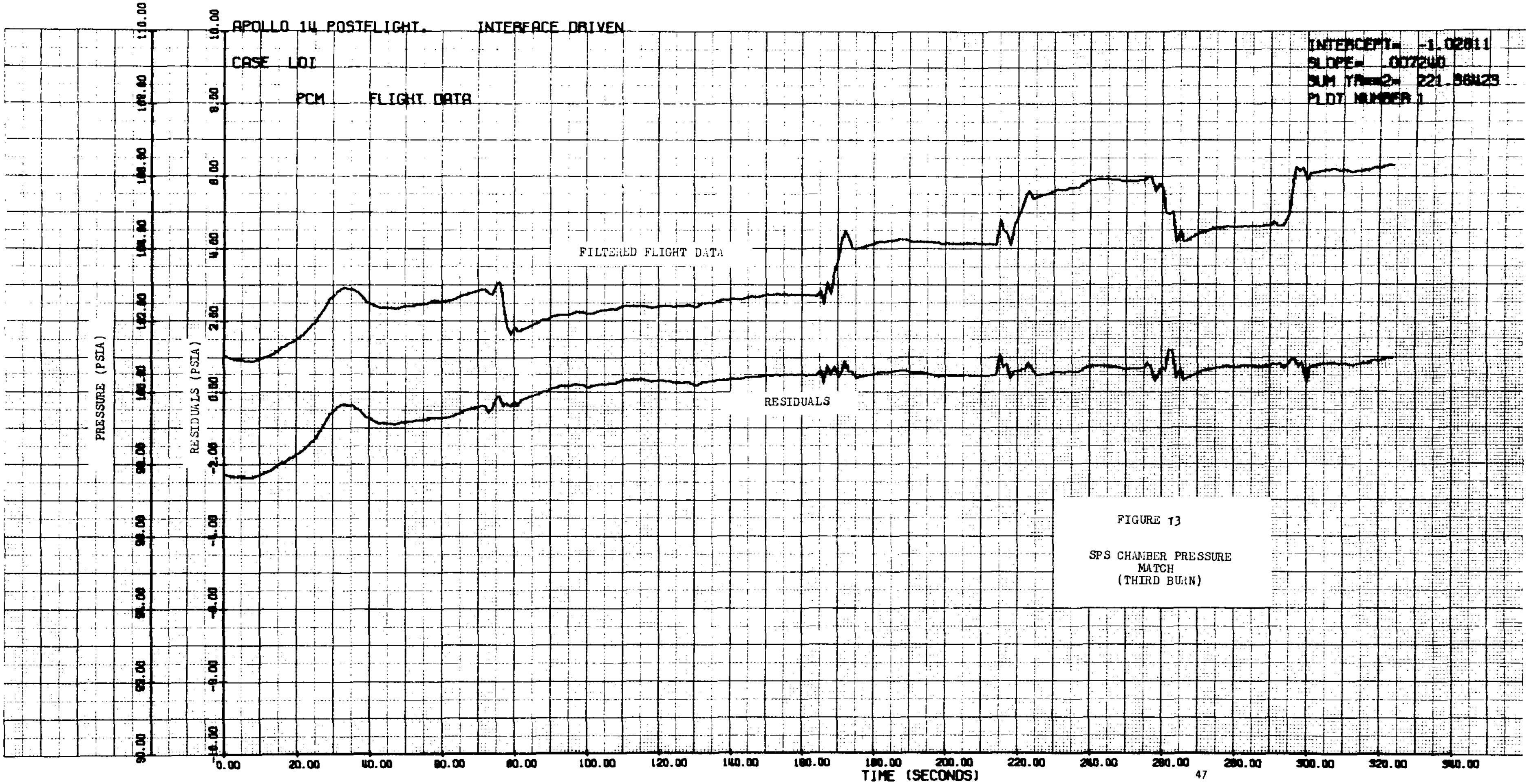


FIGURE 13

SPS CHAMBER PRESSURE MATCH (THIRD BURN)

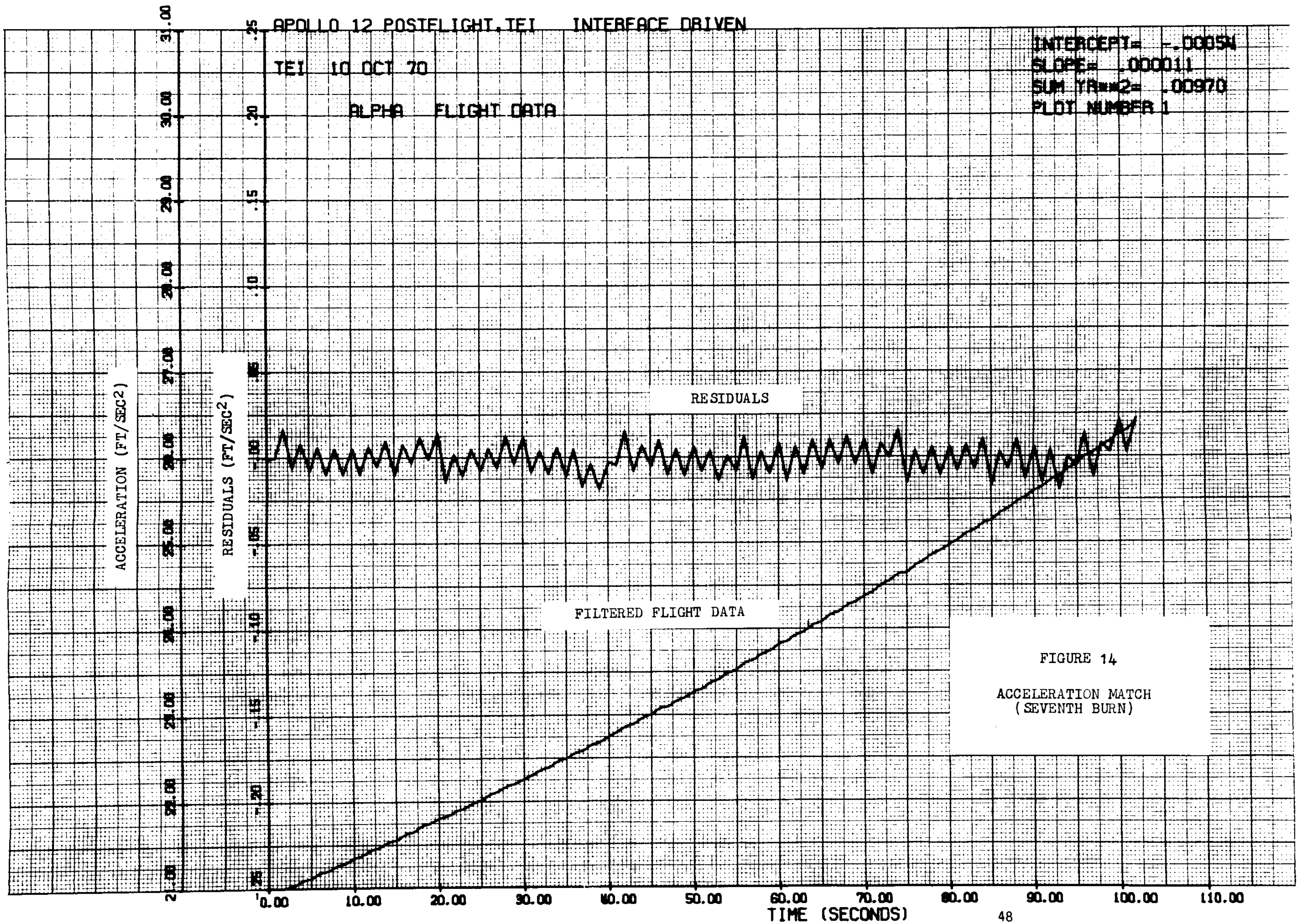


FIGURE 14

ACCELERATION MATCH
(SEVENTH BURN)

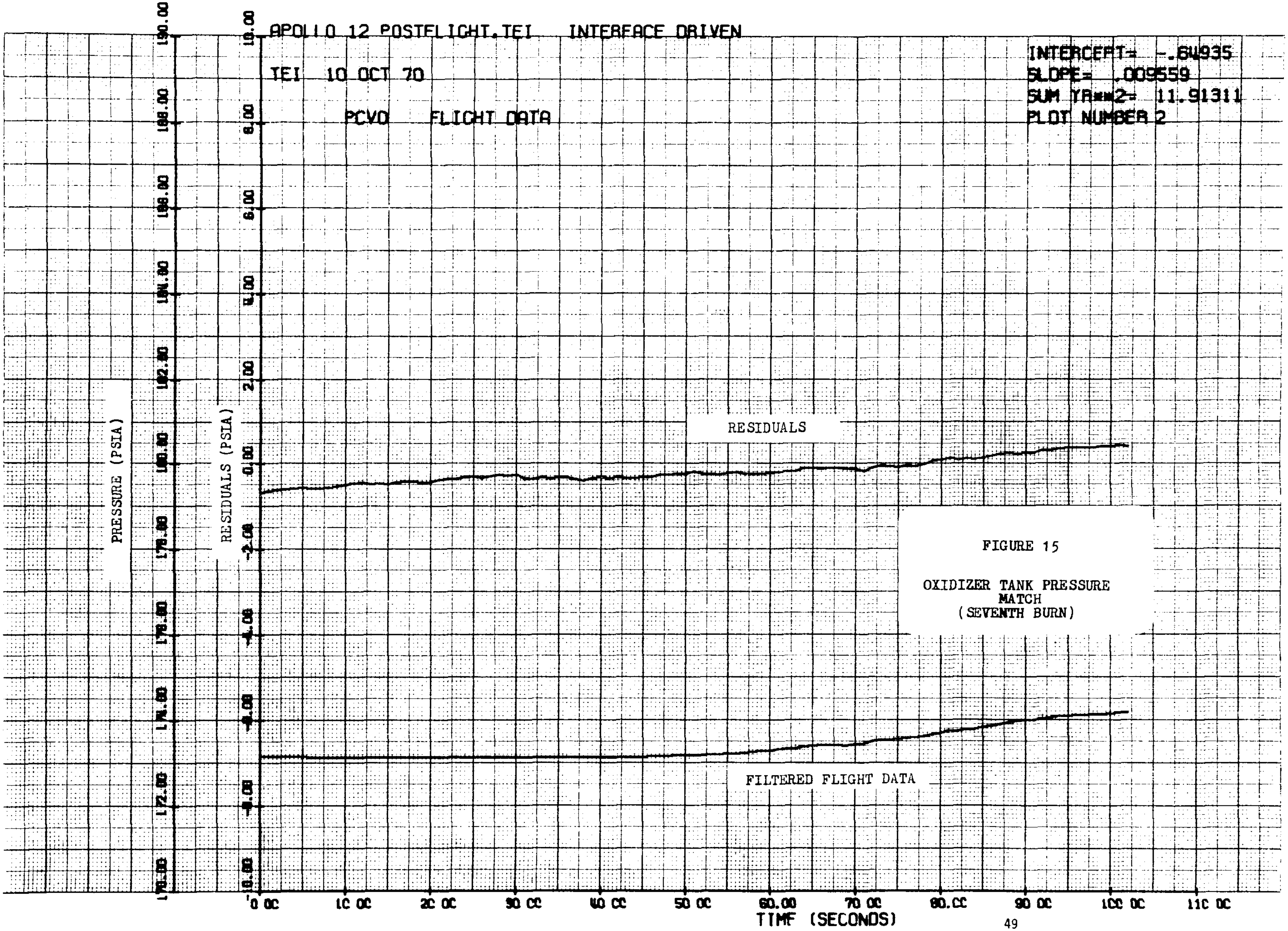


FIGURE 15

OXIDIZER TANK PRESSURE
MATCH
(SEVENTH BURN)

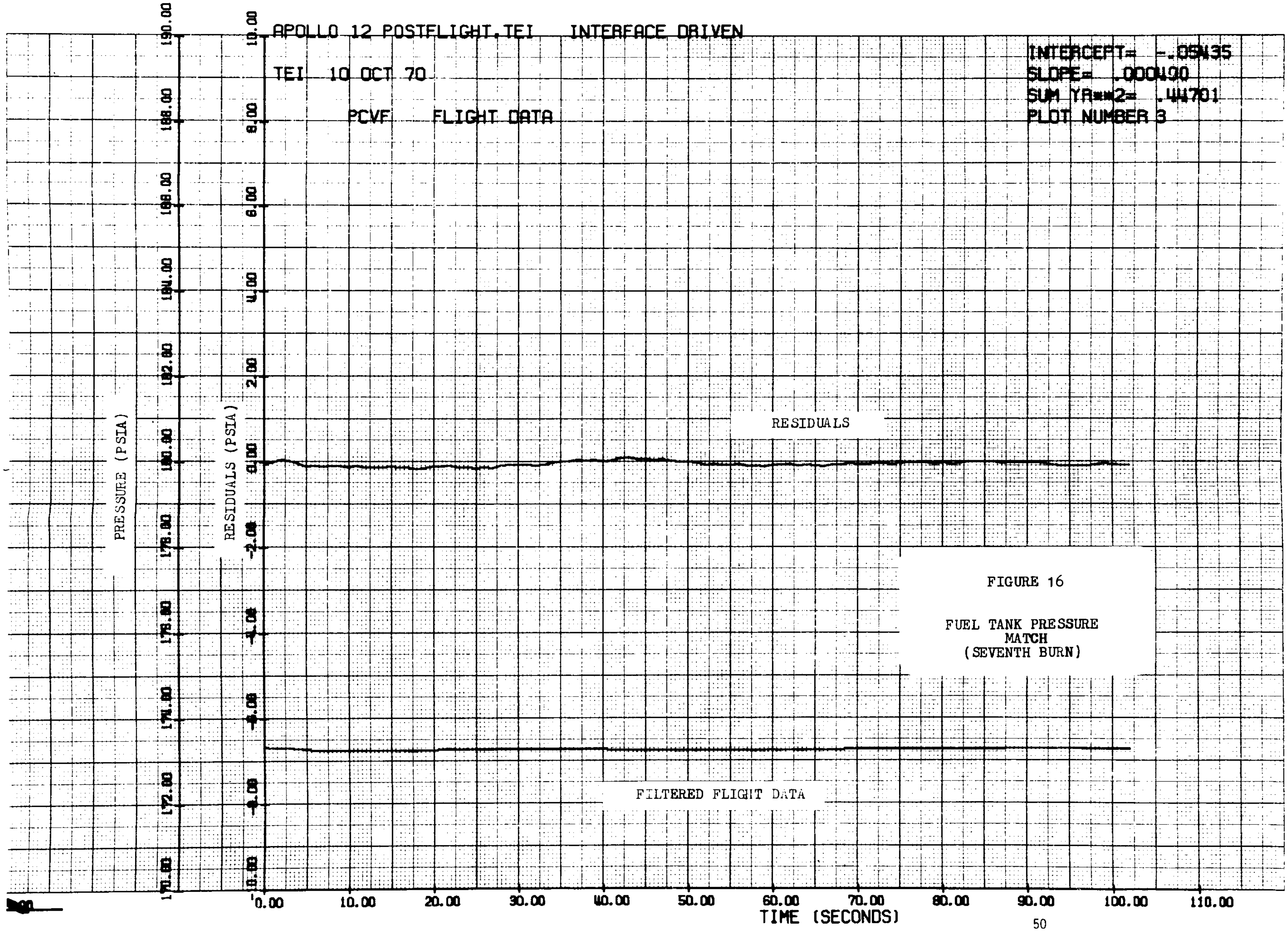


FIGURE 16

FUEL TANK PRESSURE
MATCH
(SEVENTH BURN)

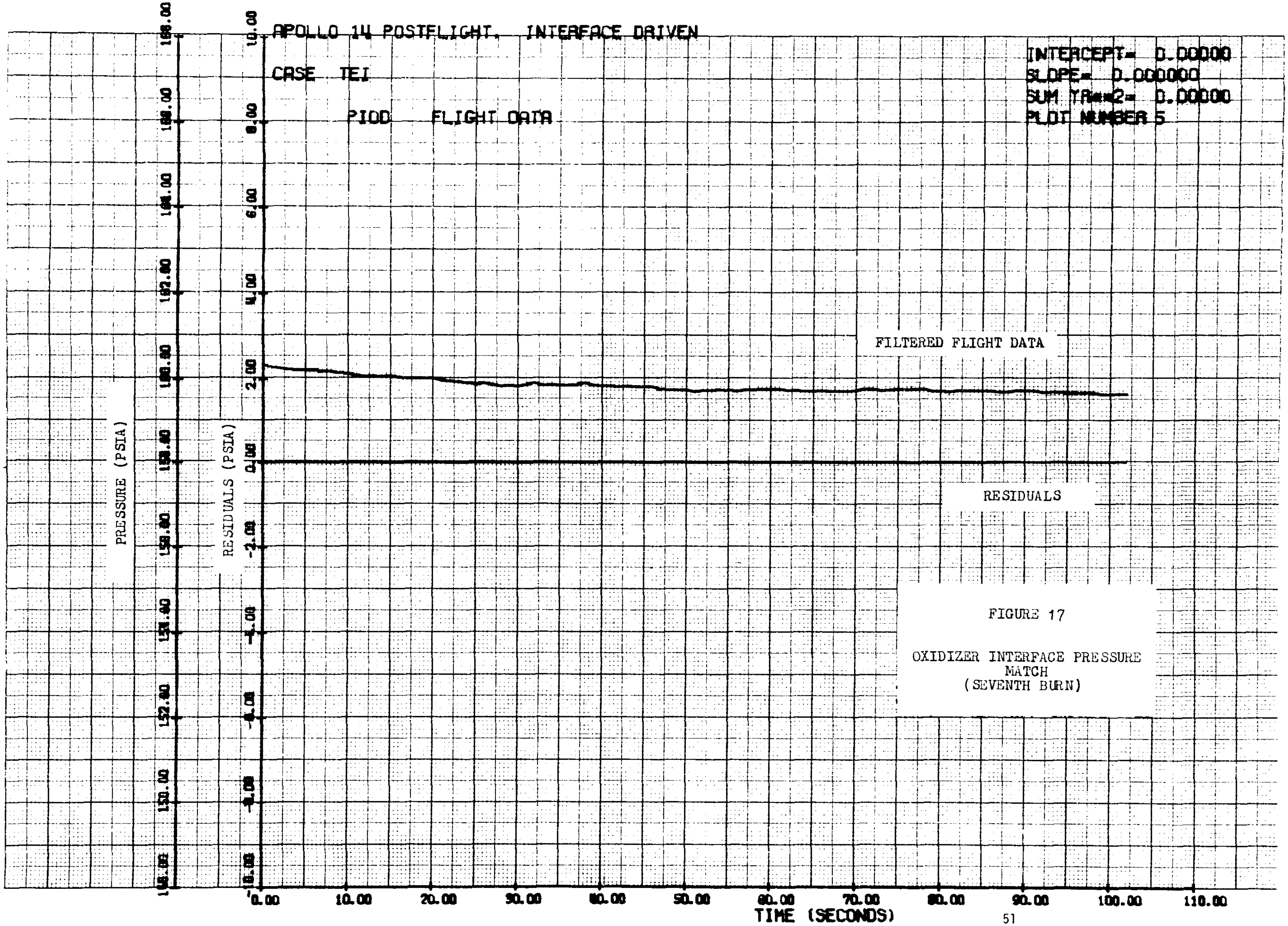


FIGURE 17

OXIDIZER INTERFACE PRESSURE MATCH (SEVENTH BURN)

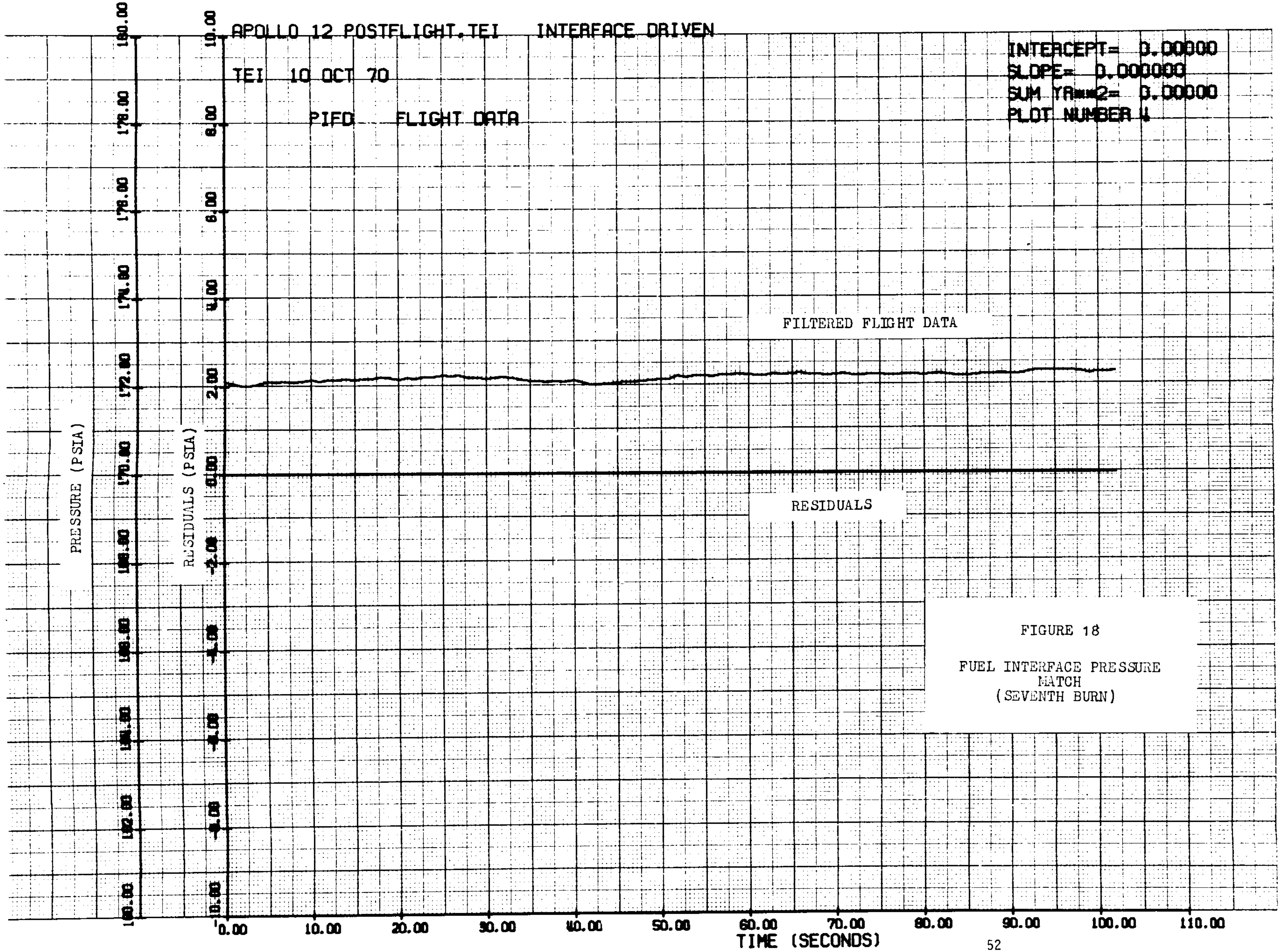
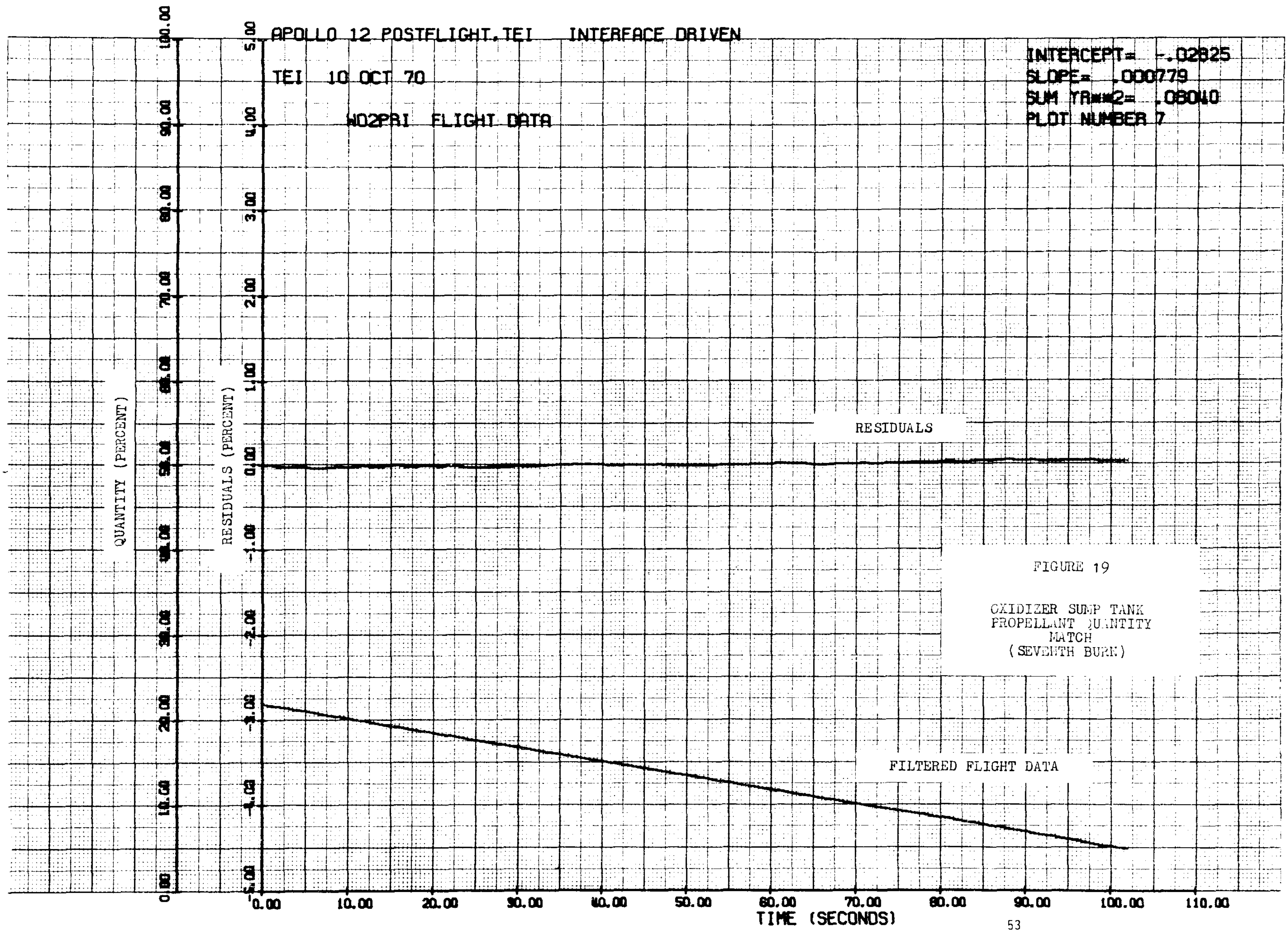


FIGURE 18

FUEL INTERFACE PRESSURE MATCH (SEVENTH BURN)



APOLLO 12 POSTFLIGHT.TEI INTERFACE DRIVEN
TEI 10 OCT 70
WO2PRI FLIGHT DATA
INTERCEPT= -.02825
SLOPE= .000779
SUM YR**2= .08040
PLOT NUMBER 7
RESIDUALS
FILTERED FLIGHT DATA
QUANTITY (PERCENT)
RESIDUALS (PERCENT)
TIME (SECONDS)
FIGURE 19
OXIDIZER SUMP TANK
PROPELLANT QUANTITY
MATCH
(SEVENTH BURN)

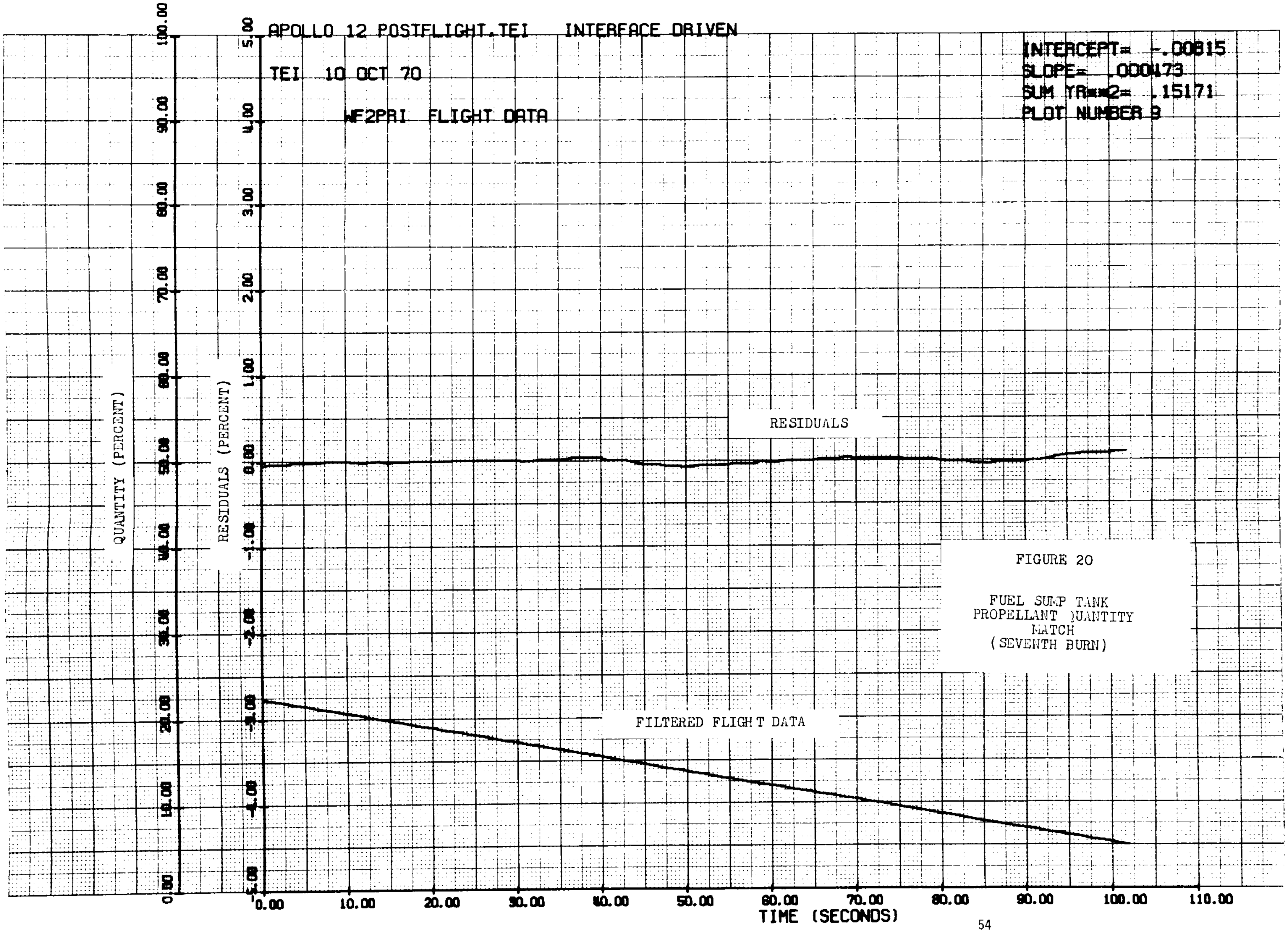


FIGURE 20

FUEL SUMP TANK
PROPELLANT QUANTITY
MATCH
(SEVENTH BURN)

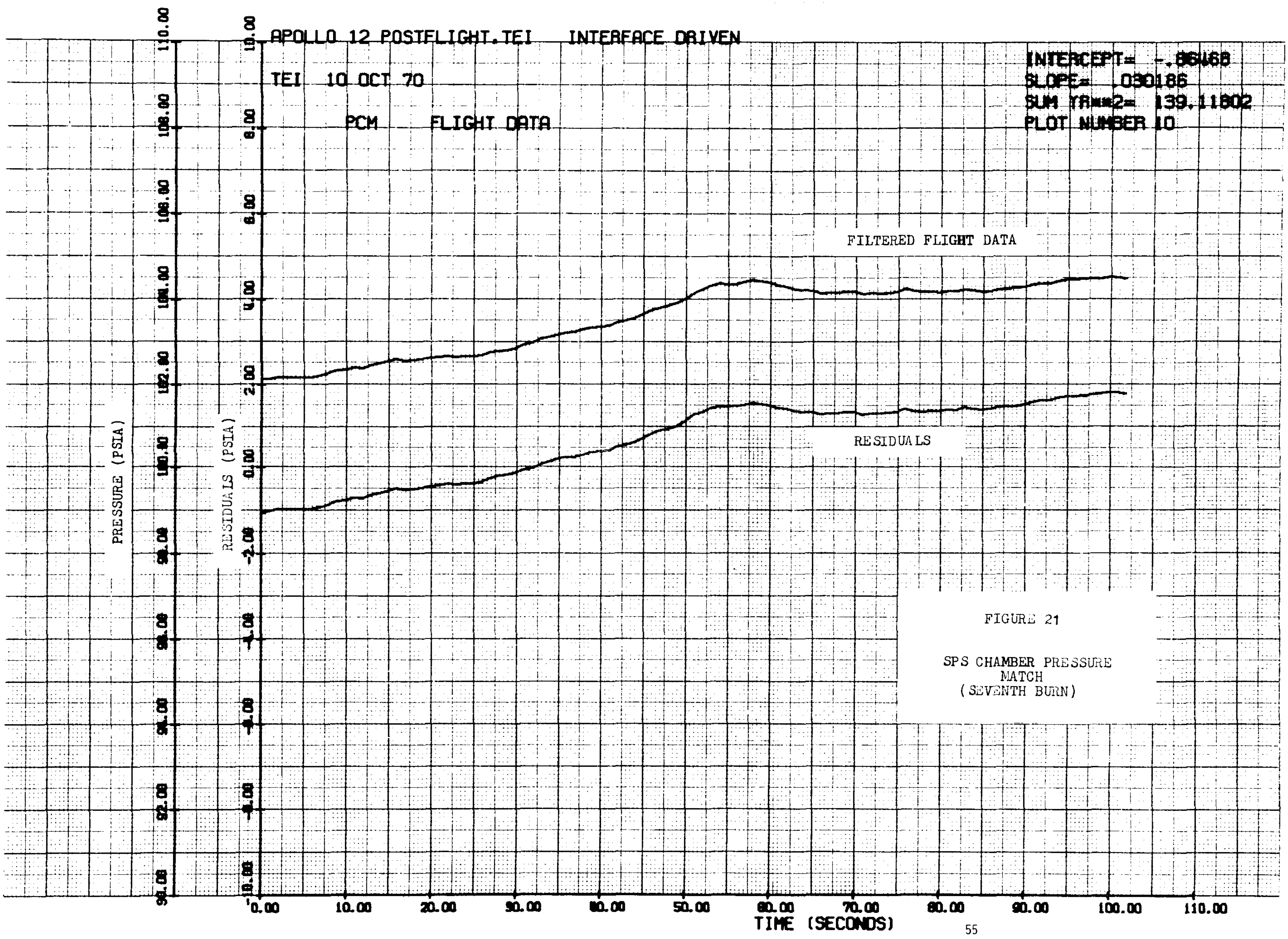
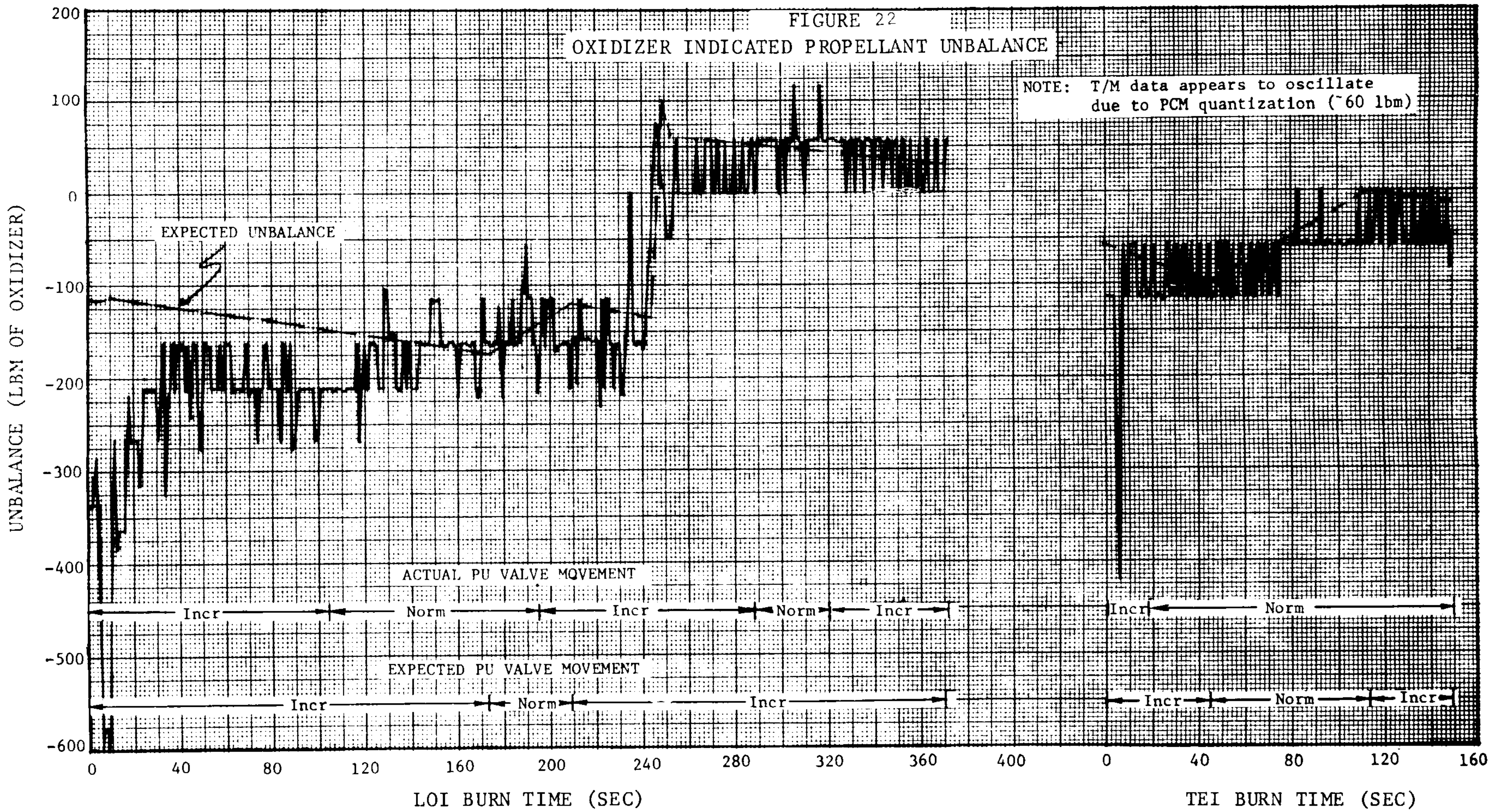


FIGURE 21

SPS CHAMBER PRESSURE MATCH (SEVENTH BURN)

FIGURE 22

OXIDIZER INDICATED PROPELLANT UNBALANCE



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